NASA CONTRACTOR REPORT

NASA CR-356

NASA CR-356

ORBITAL TESTING REQUIREMENTS FOR GUIDANCE AND CONTROL DEVICES

VOLUME II

by F. Hercules and R. Butler

Prepared under Contract No. NASw-1067 *by*
MCDONNELL AIRCRAFT CORPORATION
St. Louis, Mo.
for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • DECEMBER 1965

NASA CR-356

ORBITAL TESTING REQUIREMENTS FOR GUIDANCE AND CONTROL DEVICES

VOLUME II

By F. Hercules and R. Butler

Prepared under Contract No. NASw-1067 by
MCDONNELL AIRCRAFT CORPORATION
St. Louis, Mo.

for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

For sale by the Clearinghouse for Federal Scientific and Technical Information
Springfield, Virginia 22151 – Price $7.00

FOREWORD

This report, Volume II, is the second of two volumes reporting work accomplished under contract NASw-1067 initiated in August 1964.

This program determined what guidance and control technologies would require or could profit from orbital testing, and defines experiments which fulfill these requirements.

This volume contains the descriptions of candidate experiments. Volume I of this report summarizes the work performed on this program and describes the procedure by which the experimental selection was accomplished.

This program was conducted by personnel of the Space and Missile Electronic Systems Department of McDonnell Aircraft Corporation. The chief contributors were: R. P. Bennett, R. E. Butler, F. P. Hercules, E. H. Johnson, P. W. Jones, and P. Seligsohn.

Acknowledgment is made of the assistance of the following during the performance of this study: Prof. R. H. Cannon, Jr. and Dr. D. B. DeBra of Stanford University; Mr. N. S. Johnson and Mr. W. R. Wehrend of the NASA Ames Research Center under whose technical supervision the study was performed.

TABLE OF CONTENTS

LIST OF TABLES

LIST OF TABLES (CONT'D.)

LIST OF TABLES (CONT'D.)

LIST OF FIGURES

LIST OF FIGURES (CONT'D.)

LIST OF FIGURES (CONT'D.)

LIST OF FIGURES (CONT'D.)

LIST OF FIGURES (CONT'D.)

1.0 INTRODUCTION

Thirty experiments have been selected from a candidate list of over 100 suggested tests. The selected experiments were divided into three groups or Categories (A, B and C) according to the needs for orbital test, development status and dependence on carrier vehicle design. Volume I summarizes the suggested tests, the method used in selecting experiments and the criteria used in categorizing the experiments. This volume contains technical descriptions for the thirty experiments.

The seven Category A experiments, which cannot be adequately ground tested, have a high potential for future applications. In addition, orbital test data from similar experiments is not being obtained on another program. The eight Category B experiments also have high potential for future applications. However, orbital test data may be obtained from similar experiments on other programs;and, in some cases, ground testing can provide useful data. The fifteen Category C experiments are considered worthwhile for future study but were not fully explored on this program. The majority of these experiments require either additional development and ground test or carrier vehicle selection in order to define meaningful tests.

Performance of the selected experiments in a piggyback fashion was a primary ground rule in their design. For the successful implementation of the piggyback approach, the carrier vehicle is expected to supply a majority of the experimental support functions, although certain unique areas are to be supplied by the experimental equipment. The areas of vehicle support which were assumed available for conducting the experiments include attitude control, the master attitude reference, acquisition and tracking aids, the time reference system, the data handling system, the command receiver, and electrical power. The assumptions made regarding these vehicle support functions are described in Table 1-1 . Experiments which require

TABLE 1-1
EXPERIMENT SUPPORT FUNCTIONS

EXPERIMENT SUPPORT FUNCTION	ASSUMED VEHICLE SUPPORT
Attitude Control	Maintain the required orientation, attitude accuracy and stability.
Master Attitude Reference	Supply any required attitude reference.
Acquisition and Tracking Aids	Supply all acquisition and tracking aids required to perform the experiment.
Time Reference System	Supply the time reference required to program and time the experimental functions.
Data Handling System	Supply all equipment required to receive the outputs from the experiment signal conditioners, and multiplex, encode, store and transmit them.
Command Receiver	Supply the equipment which will receive and decode ground commands to the experiment. Subcoding of multiple commands on one wire will be used only if ground command requirements are excessive.
Electrical Power	Electrical power is 28 ± 4 VDC.

housekeeping data from the vehicle support equipment have access to the data as required. With the exception of attitude control, any incompatibilities between the carrier vehicle capabilities and the experiment requirements can be met by incorporating additional supporting equipment into the experiment packages. However, attitude control compatibility must be achieved.

The programmer, signal conditioner, and power supply represent unique support equipment which must be tailored to meet the specific requirements of each experiment. Programmer functions include sequencing the experiment and measuring elapsed time when required. The signal conditioner converts the experimental data to a format which is acceptable to the data handling system. The power supply converts the vehicle 28 VDC power to that required by the experiment and also includes all required filtering and RFI networks.

For each experiment, there usually are several methods of conducting the orbital test. In most cases, these alternate test methods are discussed in the background section of each experiment. A single test method was selected for detailed discussion so that vehicle constraints could be defined. The test method was selected on the basis of achieving the accuracy goals while using state-of-the-art measurement instrumentation and keeping the experiments as simple as possible. Following carrier vehicle selection based on the derived constraints or other considerations, further definition of the recommended test method and a detailed error analysis are required. The experiment descriptions identify the major error sources but do not contain a detailed error analysis.

The size, weight and power estimates for experiment equipment are based on presently available packaging configurations. However, since much of the experiment equipment is mechanical in nature, the use of microelectronic techniques will not yield an overall weight and volume reduction proportional to the 50 to 75 percent savings in electronics. The development plan for each experiment is based on engineering estimates.

To facilitate comparison between experiments, an outline generally applicable to all experiments was used in preparing the technical descriptions. The outline used for the Category A and B experiment descriptions includes the eight sections described in Table 1-2. To avoid the repetition of a lengthy discussion on master attitude reference and data handling systems in each experiment, these two subjects are discussed in the appendices. Only the unique attitude reference and data handling problem areas are discussed in each experiment description. For each Category C experiment, the outline includes the objective, background and a discussion of suggested tests.

TABLE 1-2

EXPERIMENT DESCRIPTION FORMAT

SECTION	PURPOSE
Objectives	• Describe goals to be met • State uses for experimental results
Background	• Define the test to be accomplished • Discuss applicable theory • Determine need for space tests • Describe devices, techniques, or concepts capable of evaluating the test • State reasons for choosing a particular approach
Functional Description	• Describe functional performance of experimental equipment • Describe test method • Discuss major error sources
Experiment Physical Parameters	• Define the size, weight, and power requirements of the experimental equipment • Define these parameters for the programmer, signal conditioner, and power supply
Data Parameters	• Define the following parameters for all major experimental data points: signal format and frequency, physical range, accuracy, and desired sampling rate
Vehicle Orbit and Attitude Requirements	• Define vehicle orbit parameters: altitude, inclination, ellipticity • Define vehicle attitude control parameters: orientation, pointing accuracy, stabilization
Experiment Support Data Handling Requirements	• Define experimental support requirements: signal conditioning, programming, master reference, environmental control, mounting, total mission power, ground commands • Define data handling requirements
Experiment Flight Test Plan	• Describe experiment development plan which will provide a time schedule for experiment design, development and fabrication, system integration, etc. • Define a flight test sequence which will provide a time-based sequence of events describing the experiment performance • Describe the approach to post-flight data reduction • Discuss possible future tests

2. CATEGORY A EXPERIMENTS

2.1 Electro Static Gyro

2.1.1 Objective - The Electro Static Gyro (ESG) tests are intended to demonstrate its projected high accuracy in a space environment and to investigate the operational problems associated with using the gyro in a space vehicle. Specifically, the objectives of the tests are to:

a. Measure the drift performance of the ESG in a near zero-g environment.

b. Demonstrate the operation of a three mode (high-g, 0.5 g and near zero-g) suspension system.

c. Demonstrate the feasibility of remote start-up and orientation of an ESG.

2.1.2 Background - The ESG is a modern gyro concept which has received extensive design and development effort in the past five years. Thousands of hours of testing in both laboratory and earth operational environments have demonstrated both the practicality and excellent performance of this gyro mechanization. Models are presently being developed for applications in space vehicle attitude reference systems. Additional space uses which have been suggested include:

a. Stabilization of a gimballed accelerometer triad,

b. Stabilization of a gimballed space sextant (Reference 1), and

c. Attitude reference and/or acceleration instrumentation for a drag-free satellite (Reference 2).

The ESG is attractive for these, and other, space applications because of its potential for highly reliable performance, low drift rates, and low power consumption. Theoretically, in the near zero-g conditions of an orbital or deep space environment, the ESG should exhibit little or no drift. This is because the predominant drift torques in the gyro are proportional to either the applied acceleration or the expected maximum acceleration, i.e., the level for which the electrical suspension is preloaded.

ESG CUTAWAY

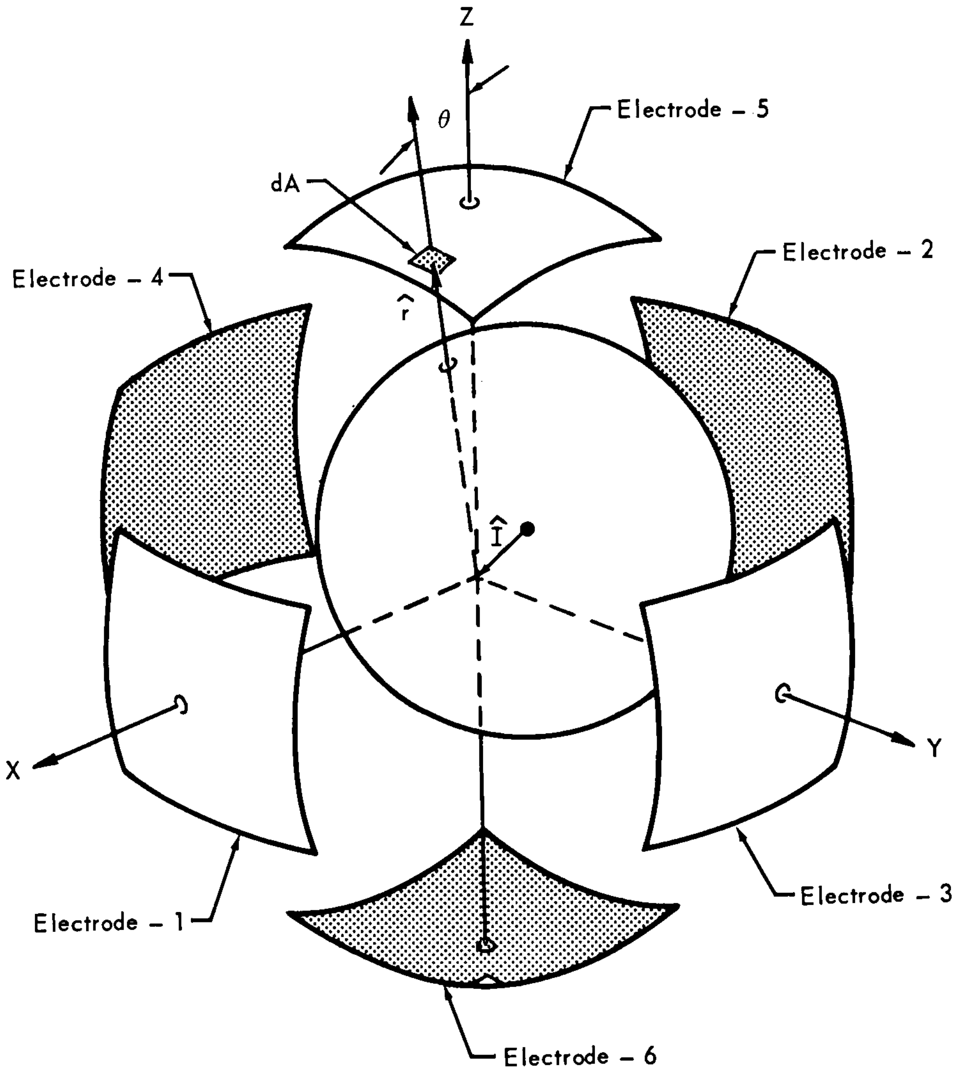


FIGURE 2.1–1

The ESG's basic simplicity is shown in the cutaway view of Figure 2.1-1. Its only moving part is the hollow spherical beryllium rotor contained inside a ceramic envelope which serves as a vacuum vessel. The rotor is suspended in the center of the envelope by electric fields. The measured capacitance between the rotor and the electrodes on the interior surface of the ceramic envelope indicates the position of the rotor and controls the suspension voltages applied to the electrodes.

After the ESG is suspended, the rotor is spun to a preselected operating speed, then torqued and damped to the desired orientation using magnetic fields generated by coils mounted externally to the ceramic envelope. After demagnetization, the rotor is allowed to coast undisturbed. Thus the ESG is a two-degree-of-freedom (free) gyro with its spin axis nominally maintained in a fixed direction in inertial space. Since the ceramic envelope's interior is maintained at a hard vacuum (10^{-8} mm Hg), and since integral shielding protects the gyro from magnetic fields, the rotor can coast for months with only negligible slowing of its speed.

Due to outgassing of materials within the envelope, it has been estimated that the vacuum level can be maintained for only thirty to sixty days without an external vacuum source. In the present Honeywell design, a small getter-ion vacuum pump is used to maintain the vacuum. Venting the envelope to space has been suggested as an alternate for maintaining the vacuum. This solution has advantages, but tests must be performed to determine the effects, if any, of contaminants such as reaction jet exhaust materials which may remain in the vicinity of the vehicle.

Rotor orientation is obtained by means of optical pickoffs mounted on the envelope. These pickoffs observe a pattern on the rotor. For "strapped-down" applications, the optical pickoff outputs directly indicate gyro spin axis orientation relative to the vehicle. For gimballed applications, the outputs of the optical pickoffs are used to control a gimbal drive system and gimbal angle readouts then indicate the gyro spin axis orientation relative to the vehicle.

The knowledge to be derived from ESG tests in an orbital assessment program can best be understood by examining the basic drift producing torques that affect the accuracy of an ESG (Reference 3). Only two sources, rotor mass unbalance and electric suspension forces, cause drift producing torques of any appreciable magnitude in the ESG. Mass unbalance drift is caused by a difference between the rotor mass center, through which the acceleration forces act, and the rotor center of support through which the effective electric suspension force acts. The magnitude of mass unbalance torque which causes drift (axial mass unbalance only) is given by:

$$\overline{T}_{mu} = m\,(\overline{1} \cdot \overline{S})\,\overline{S} \times \overline{a} \qquad (1)$$

where: m = rotor mass

$\overline{1}$ = vector from rotor surface centroid to rotor mass center

$\overline{S}$ = unit vector along rotor spin axis

$\overline{a}$ = acceleration

The drift rate of the gyro is directly proportional to this torque. The decrease in acceleration magnitude obtained by testing the gyro in an orbiting vehicle instead of an earth-bound laboratory will, of course, be directly reflected in a decrease of the spurious torque caused by mass unbalance. This zero-g test condition cannot be simulated on the ground by suitable alignment (i.e., spin axis parallel to gravity axis) since the ESG is used with its spin axis space fixed. The ESG spin axis will therefore tumble with respect to the gravity vector.

The second major ESG torque producing source - electric suspension forces - is a result of suspension force components which are normal to the radius line through the center of suspension. Corresponding differential areas of the ESG rotor and electrodes can be analyzed by comparing them to similar areas of a parallel plate capacitor of infinite area. The force exerted on the differential rotor area is therefore given by

$$d\,\overline{F} = \frac{\epsilon_o}{2} \cdot \frac{V^2}{\Delta} \cdot \overline{N}\, dA \tag{2}$$

where: $\overline{F}$ = force

V = rotor-to-electrode potential

Δ = normal rotor-to-electrode distance (gap)

$\overline{N}$ = unit vector normal to rotor surface

A = area of rotor surface

ϵ_o = permittivity of free space

The component of this force which passes through the center of suspension of the rotor supplies the rotor suspension force. The component of this force which is normal to the radius line (r) from the rotor center of suspension causes a drift torque. This drift torque is given by:

$$\overline{L}_E = \int_A r \times d\overline{F} \tag{3}$$

In equation (2), the suspension force is a function of the voltage squared factor. The suspension system is operated in a linear mode with a voltage preload adequate to support the rotor for any expected acceleration. Operation of the ESG in the quiescent conditions of an orbiting space vehicle will allow these voltage preloads to be reduced significantly, with a corresponding decrease in the spurious torques caused by the electric suspension forces. Again, the near zero-g test condition cannot be simulated in an earth-bound laboratory since the low preload voltages will not provide rotor suspension in a one-g environment.

Additional disturbance torques are caused by gas drag and magnetic fields. These effects are neglected here since they are assumed to be controlled to

ESG EXPERIMENT FUNCTIONAL BLOCK DIAGRAM

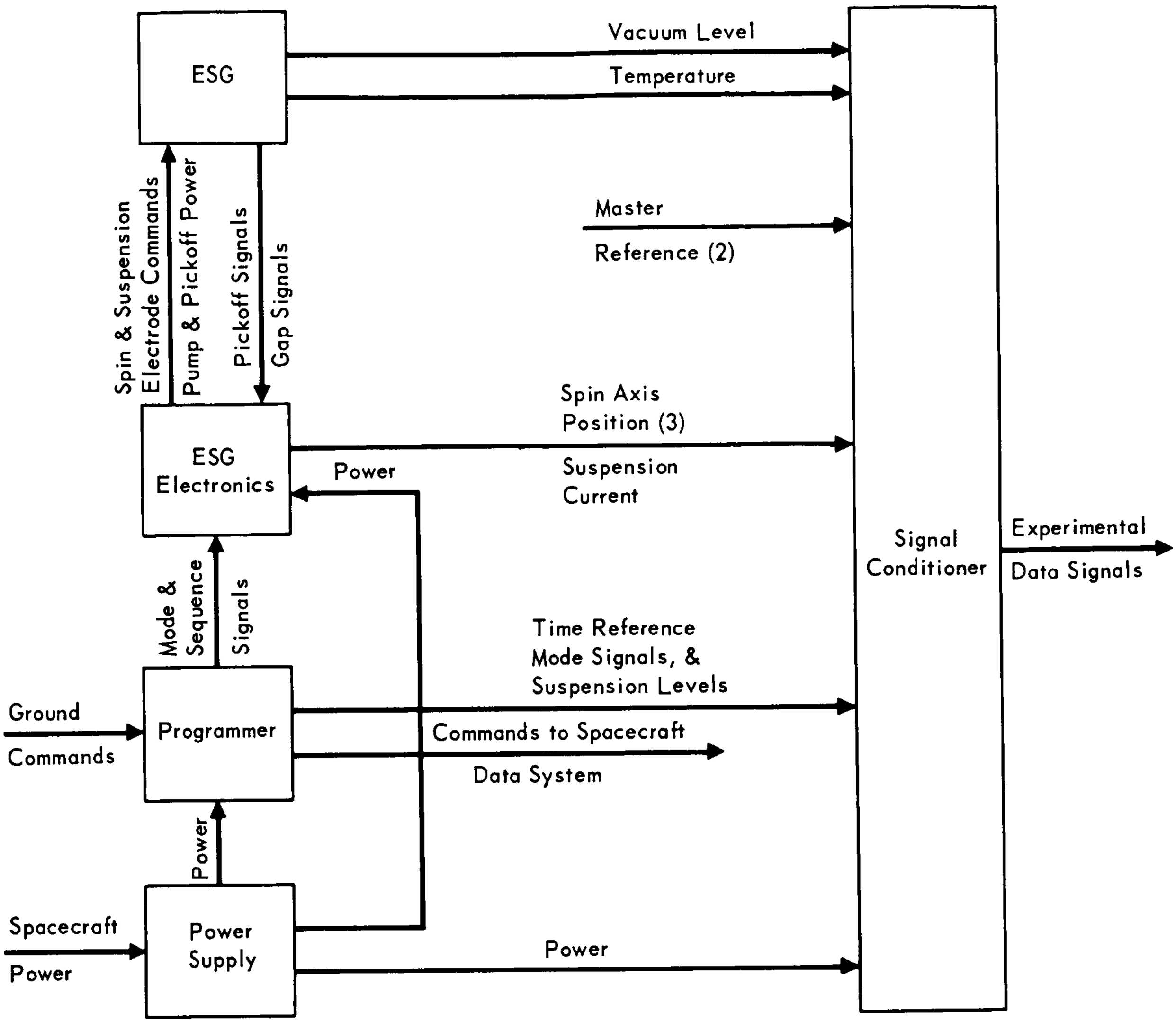


FIGURE 2.1-2

negligible levels by proper design, i.e., maintain vacuum and use magnetic shielding.

2.1.3 Functional Description - The ESG experiment is functionally in accordance with the diagram of Figure 2.1-2. Experiment equipment consists of a strapped-down ESG and the ESG electronics. For drift measurements, a master attitude reference and a time reference are required. In the discussion of test method, the master reference is assumed to be a star tracker.

Experiment Equipment - The strapped-down ESG is used since the experiment is less complicated and applicable equipment is more readily available than the gimballed configuration. However, the gimballed ESG requires only 14 bits of readout data to provide one arc second angular accuracy over its limited travel, whereas 20 bits are required to provide equivalent accuracy over the 360 degree range of the strapdown configuration. If the ESG is operated on an inertially oriented vehicle, the vehicle would act as a gimbal system and the gimballed ESG configuration could be used. This would relieve the data handling problem concerning the readout, but this configuration could not provide a test during launch. The strapped-down ESG consists of the rotor with its associated suspension and torque electrodes, the pickoffs and the vacuum pump. The ESG electronics is basically two sections: Remote Start-Up Electronics and Pickoff and Suspension Electronics. The former provides the required signals to spin up, orient and damp the rotor. The latter comprises the circuits which provide desired suspension level, control rotor suspended position, and transform the ESG optical pickoff signals into binary digital signals. The suspension level can be operated in three modes: Mode 1 maintains suspension through the high-g launch environment; Mode 2 is preloaded for a 0.5 g level; and Mode 3 maintains suspension under the near zero-g levels in orbit by using the lowest suspension forces possible. Although three particular modes have been chosen, additional ones could be used to give a better definition of drift in terms of suspension level. These three were chosen because they represent likely operational modes, and in addition, will provide data on the major torque producing mechanisms.

Test Method - The gyro is energized prior to launch and operates through launch. Following vehicle stabilization in orbit and star tracker acquisition of the guide star, the ESG spin axis orientation relative to the gyro case reference is recorded. Also, gyro case orientation relative to the known guide star is recorded. The above orientation measurements, combined with knowledge of the ESG spin axis orientation prior to launch, are used to estimate gyro drift during launch. The ESG is then automatically sequenced to operate in suspension Mode 2, at 0.5 g level, and the drift performance tests commence. The information to be obtained from the test is both short and long term ESG drift characteristics under various suspension levels. The drift is measured by periodically reading out and recording the gyro spin axis orientation relative to known guide stars. A time label is given to each data point to allow calculation of ESG drift rates on the ground. For this initial test, the duration is one week to establish both short and long term drift trends.

After collecting data samples for one week, the suspension system is changed to Mode 3, the near zero-g level. Drift measurements over short and long time periods are recorded as above for a period of one week. The data is then compared with ground tests at one-g and the previous orbit test at 0.5 g suspension level to obtain three points on a plot of drift rate versus suspension level. The data can then be extrapolated to other suspension levels, e.g., launch levels, and estimated performance compared with actual performance through launch.

The next significant procedure is to de-energize the ESG in preparation for remote start-up. The de-energizing sequence is initiated by ground command. The first and most critical step of the sequence is to de-spin the rotor by applying a DC voltage to the spin-up coils. Following de-spin, all power is removed from the suspension coils, vacuum pump, and readout device. After remaining de-energized for some time period, the remote starting sequence is initiated by ground command. The functional sequence for remote start-up is as follows:

a. evacuate chamber (if necessary),

b. suspend rotor at the 0.5 g level,

c. spin-up rotor,

d. torque to preferred orientation, and

e. damp to orientation.

These functions are sequenced automatically following the initial command signal. Additional drift performance tests are now conducted in a fashion similar to those described earlier.

Further tests will depend on vehicle life and power capability. Possible tests include a remote start-up test with a near zero-g suspension level, drift testing with the vacuum pump off to demonstrate possible degradation due to outgassing of materials into the envelope, and drift tests with the envelope vented to space to demonstrate possible degradation from contaminants in the region of the vent.

Major Error Sources - For evaluating ESG drift rate performance, the major error sources are ESG and master reference pickoff resolution, time reference accuracy, pickoff time constant differences, and short term structural warping between ESG and master reference mounting bases. ESG pickoff resolution can be on the order of 1 arc second while the timer accuracy can be on the order of 1 second in an 8 hour period. The remaining error sources are a function of carrier vehicle, mounting design, and master reference selection; however, 20 arc seconds appears to be a reasonable error budget for all error sources.

2.1.4 Experiment Physical Parameters - As stated previously, the strapped-down ESG is recommended for the initial orbital test. Table 2.1-1 summarizes the physical parameters for the primary subsystem which includes the ESG and supporting equipment. The ESG electronics is broken into its two sections and the vacuum pump is specified separately. In the event that the remote spin-up is eliminated or a different means of vacuum maintenance is employed, the physical parameters

TABLE 2.1-1

ESG EXPERIMENT PHYSICAL PARAMETERS

	SIZE		WEIGHT (LB.)	POWER	
	VOLUME (FT.^{3})	L – W – H (IN.)		PEAK (WATTS)	NOMINAL (WATTS)
EXPERIMENT					
Electrostatic Gyro	.038	5D (Sphere)	5.0		
Vacuum Pump	.001	1.5 x 1 x 1	.3	1*	1*
Pickoff & Suspension Electronics	.208	10 x 6 x 6	10.0	10*	5*
Remote Start-up Electronics	.028	4 x 4 x 3	1.7	5.5*	
SUPPORT					
Programmer	.019	4 x 4 x 2	1.0	2*	1*
Signal Conditioner	.002	3 x 1 x 1	.1	2*	1*
Power Supply	.024	4 x 3 x 3½	2.0	25	10
TOTAL	.32		20.1	25	10

*Supplied by Power Supply

may be adjusted. The power supply includes an inverter and auxiliary batteries with the batteries being used to suspend the ESG in the event of primary power system interruption. The power supply estimates in Table 2.1-1 are based on a space vehicle supply of 28 volts DC.

2.1.5 <u>Data Parameters</u> - The parameters that are measured to properly evaluate the experiment are summarized in Table 2.1-2. The table lists first those parameters considered essential for evaluating the experiment and also lists additional parameters that will indicate equipment operating and environmental conditions. The "additional parameter" list is representative of the desired measurements and is not intended to be complete. In fact, the length of the list may change following hardware procurement and actual experiment integration into the space vehicle. The only data parameters which require special consideration are the rotor position pickoffs. These are digital pickoffs with a 360 degree angular capacity. The

TABLE 2.1–2

ESG EXPERIMENT DATA PARAMETERS

DATA POINT	PRIORITY	SIGNAL FORMAT AND FREQUENCY	PARAMETER RANGE	SAMPLE RATE (/SEC.)	ACCURACY	RECORD	ESSEN-TIAL	DESIR-ABLE
ESG Pickoff – No. 1	1					X	X	
– No. 2	1	Digital	20 Bits	1	± 1 Bit	X	X	
– No. 3	1					X	X	
Suspension Current – No. 1	1					X	X	
– No. 2	1	Analog, DC		2	1%	X	X	
– No. 3	1					X	X	
Master Reference P/O – No. 1	1	Digital	10 Bits	1	± 1 Bit	X	X	
– No. 2	1					X	X	
ESG Pressure	2			0.5		X		X
ESG Temperature	2					X		X
ESG Spin-up Power	2					X		X
ESG Torque Power	2					X		X
ESG Damp Power	2	Analog, DC	On/Off	0.2		X		X
ESG De-spin Power	2					X		X
Master Reference Power	2					X		X
Suspension – Mode 1	2					X		X
– Mode 2	2					X		X
– Mode 3	2					X		X
Time Interval	1	Digital	1 Week	0.5	± 1 Sec.	X	X	
Tolerance Exceeded	2	Analog, DC	On/Off	0.2		X		X

Note: All data points are monitored during the operation of the spin electrodes, at 90 minute intervals and during periods when one or more operating parameters has exceeded its tolerance. The operating parameters are suspension current, ESG pressure, and ESG temperature.

accuracy of these units is a measure of the number of digital bits used. Twenty bits will give an accuracy of 1.05 arc seconds. The required accuracy will be determined by the expected drift rates and the accuracy of the master reference.

2.1.6 Vehicle Orbit and Attitude Requirements - The ESG experiment places no special requirements on orbit parameters. Attitude requirements can be stated in terms of preferred orientation, attitude deviations relative to the preferred orientation, and permissible attitude rates. An inertial orientation with three-axis vehicle stabilization is preferred so that a non-gimballed master reference can be used for measuring ESG drift. The permissible attitude deviation from the preferred orientation is primarily a function of the master attitude reference sensor field of view. Since the strapped-down ESG is used, the ESG readout imposes essentially no attitude restrictions on the vehicle. However, a stable orientation is required during remote ESG start-up. If a star sensor master reference is used and has a $\pm$ 2 degree field of view, the vehicle attitudes should not be allowed to deviate over + 1 degree from the star line-of-sight to insure that the star remains in the sensor field of view. The permissible attitude rates are related to the star sensor and ESG readout time constants and to restrictions imposed by remote ESG start up. For proper remote start-up, the vehicle rates should be less than 0.05 degrees/second. Assuming a 0.01 second ESG-master reference time constant difference, the maximum readout error due to a vehicle rate of 0.05 degrees/second is 0.0005 degrees or about 1.8 arc seconds. This is considered a permissible error. Hence, vehicle rates of 0.05 degrees/second or less are acceptable provided the ESG-master reference time constant difference is 0.1 second or less.

2.1.7 Experiment Support and Data Handling Requirements - The data handling requirements discussed in Appendix B are applicable. Based on a 30 day test cycle, the power requirements defined in Table 2.1-1 and the flight test sequence in Figure 2.1-4, the total energy required per cycle is approximately 6800 watt-hours. Although no problems are anticipated in the design of the signal conditioner, unique problems do exist in the experimental support areas discussed below.

Programmer - Programmer design is based on the flight test sequence described in Figure 2.1-4. One area of the programmer required special design consideration. This is the circuitry which will monitor the ESG operational parameters: suspension current, ESG temperature and ESG pressure. If any one of these is out-of-tolerance, the programmer must command the vehicle data system to record all parameters as long as this condition exists. The operational parameters could indicate an approaching failure. Data recorded during this time would be very important to experiment analysis. The programmer must include a time reference system which is accurate to one second within an eight hour time period. The system is reset each eight hours upon ground command. This accuracy is required to support the measurement of short term drift rates. The ninety minute intervals between drift data monitoring points will be controlled by a time interval unit within five minutes accuracy.

Master Reference - The primary function of the master reference is to provide a readout, preferably digital, which indicates the angular orientation of the vehicle with respect to a known reference. Successive values of this data when compared to the readout indicating ESG spin axis orientation with respect to the vehicle, are used to determine ESG drift. Various combinations of the techniques discussed in Appendix A can be used. However, certain techniques, as discussed below, are preferred because the experiment is simplified.

If the vehicle is inertially oriented, a non-gimballed star sensor can be used as a master reference, providing that the ESG spin-axis is precisely aligned along the sensor line-of-sight prior to the start of the drift test. Alternately, two non-gimballed star sensors can be used to provide a three axis attitude reference which requires no ESG spin-axis alignment. In a similar fashion, a precise sun-sensor mounted on a solar oriented vehicle can provide the required attitude reference. Here again, the ESG spin-axis must be aligned to the sensor line-of-sight prior to starting a drift test. The alignment requirement may be eliminated by obtaining a third attitude reference axis by means of a star sensor/tracker.

For earth-oriented vehicles, the master reference can be either a gimballed star tracker or mapper, or a sun sensor/tracker in conjunction with the horizon sensor, or it may be the horizon sensor itself. However, the experiment procedure will become complicated by the requirement for star identification techniques and/or navigation data which must be incorporated into the data reduction process.

The simplest reference which may be used with earth-oriented vehicles is the horizon sensor. The vertical reference established by the sensor, coupled with navigation data, will provide a two axis attitude reference required to determine ESG drift. However, as with any of the possible reference techniques, accuracy of the experiment will be based on both sensor accuracy and navigation data accuracy. Even if the horizon sensor can achieve a 0.1 degree accuracy, this is still quite large when compared to the expected ESG drift. While addition of a star tracker will improve the sensor accuracy, the complication to the experiment equipment and procedures, both in the vehicle and on the ground, will greatly reduce experiment reliability. On the other hand, if an inertially or solar-oriented vehicle is used, navigation data will not be required and these sensors will provide the same accuracy as they would on an earth-oriented vehicle, but without the complicated gimbal structures and, to some extent, the need for star recognition techniques will be eliminated.

For overall simplicity in both experimental equipment and procedures, dual star sensors on an inertially oriented vehicle or sun/star sensors on a solar-oriented vehicle are the preferred techniques for use with the ESG experiment. Hence, either an inertial or solar orientation is preferred. The observatories such as OAO, OSO, AOSO appear to be attractive candidates for ESG testing since they are controlled to a stellar or solar orientation and incorporate either star or sun sensors or both. Should it prove impossible to perform the ESG experiment on one of the planned observatories, it may be possible to use modified observatory sensors for the master reference instrumentation on the experiment vehicle.

Environmental Control - A pressurized compartment is not required for the experiment equipment. However, the experimental package is known to be temperature sensitive. For example, the quoted starting time of 1 to 1.5 hours is applicable only in the temperature range of approximately 30 to 80°F. Further study is needed to fully define the effects.

Mounting - Since ESG drift is determined by successive measurements of spin-axis position, the original angular bias errors will be eliminated during data reduction. Thus, the mounting configuration should be designed to minimize the structural creeps and shifts, due to thermal distortion and day/night orbit conditions, which would effect the relative angular relationship between the ESG and the master reference.

Ground Commands - The experiment package will operate in conjunction with seven ground commands. These will control starting and stopping phases of ESG operation, control suspension level, reset the master timer, and control data readout. Table 2.1-3 defines the commands.

2.1.8 Experiment Flight Test Plan - The development plan for the experiment is shown in Figure 2.1-3. The gating item is the twelve month time required for

ESG EXPERIMENT DEVELOPMENT PLAN

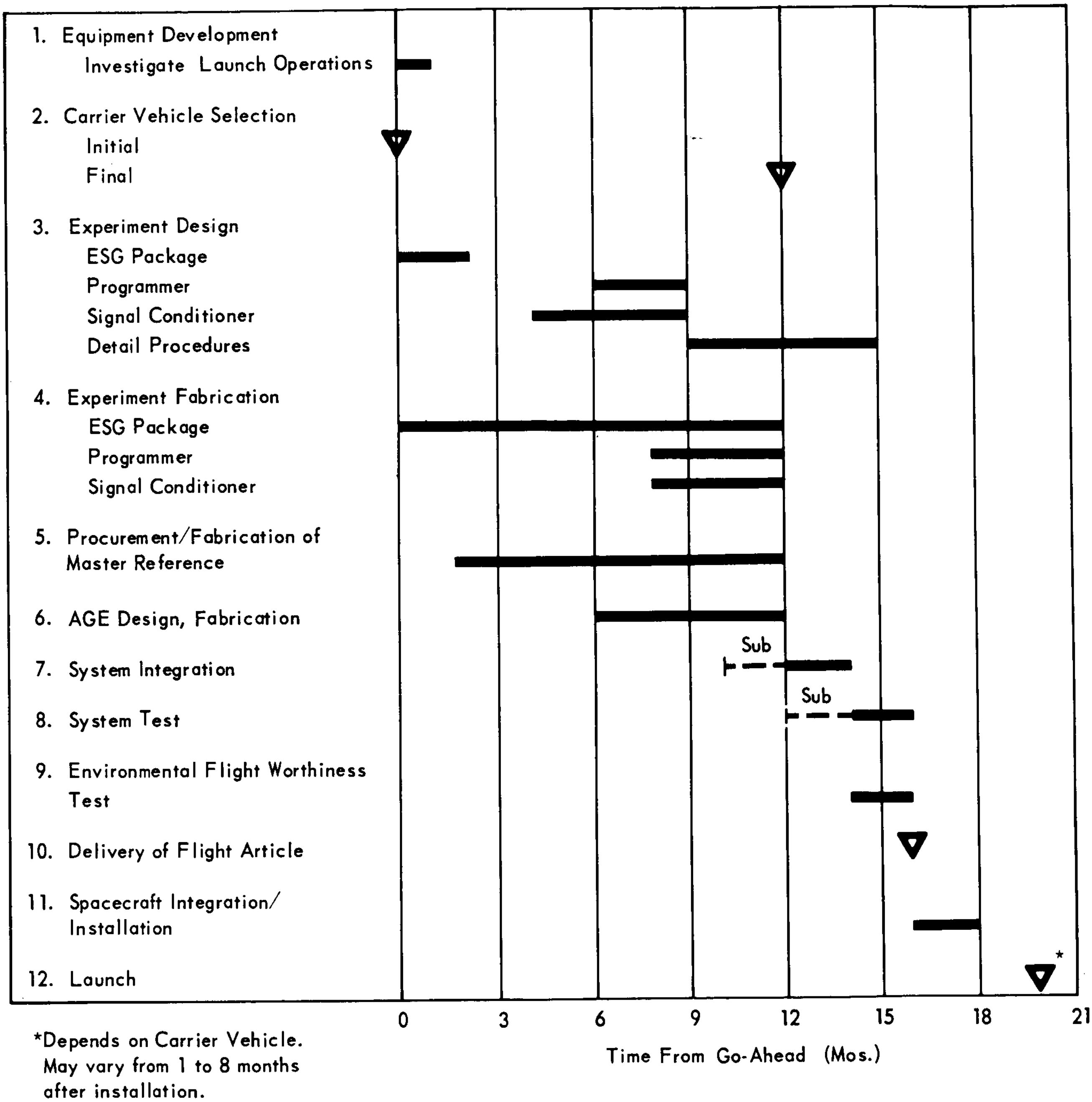


FIGURE 2.1-3

TABLE 2.1-3
ESG EXPERIMENT GROUND COMMANDS

GROUND COMMAND	ACTION
De-spin	Commands programmed sequence to de-spin, and shut down experimental equipment. Resets remote start-up sequence.
Remote Start-up	Commands programmed sequence to evacuate envelope, suspend sphere, spin-up rotor, orient and damp out nutations. Suspension mode signal must be supplied separately. Resets de-spin sequence. Commences monitoring data at 90 minute intervals.
Mode 2	Commands Mode 2 suspension level until next suspension command is received.
Mode 3	Commands Mode 3 suspension level until next suspension command is received.
Data Dump	Commands transmission of stored data.
Monitor	Commands start or stop of real time data transmission.
Timer Reset	Commands the master timer to reset to zero.

the fabrication of the ESG and its electronics and the time required to design the launch suspension system. The flight test sequence is defined in Figure 2.1-4. The sequence includes the following steps:

Prelaunch

a. Start vacuum pump.

b. Suspend rotor using mode 1 suspension level.

c. Spin rotor to full speed, orient to launch position and damp nutations.

d. Monitor experiment throughout steps (a) through (c).

Orbit

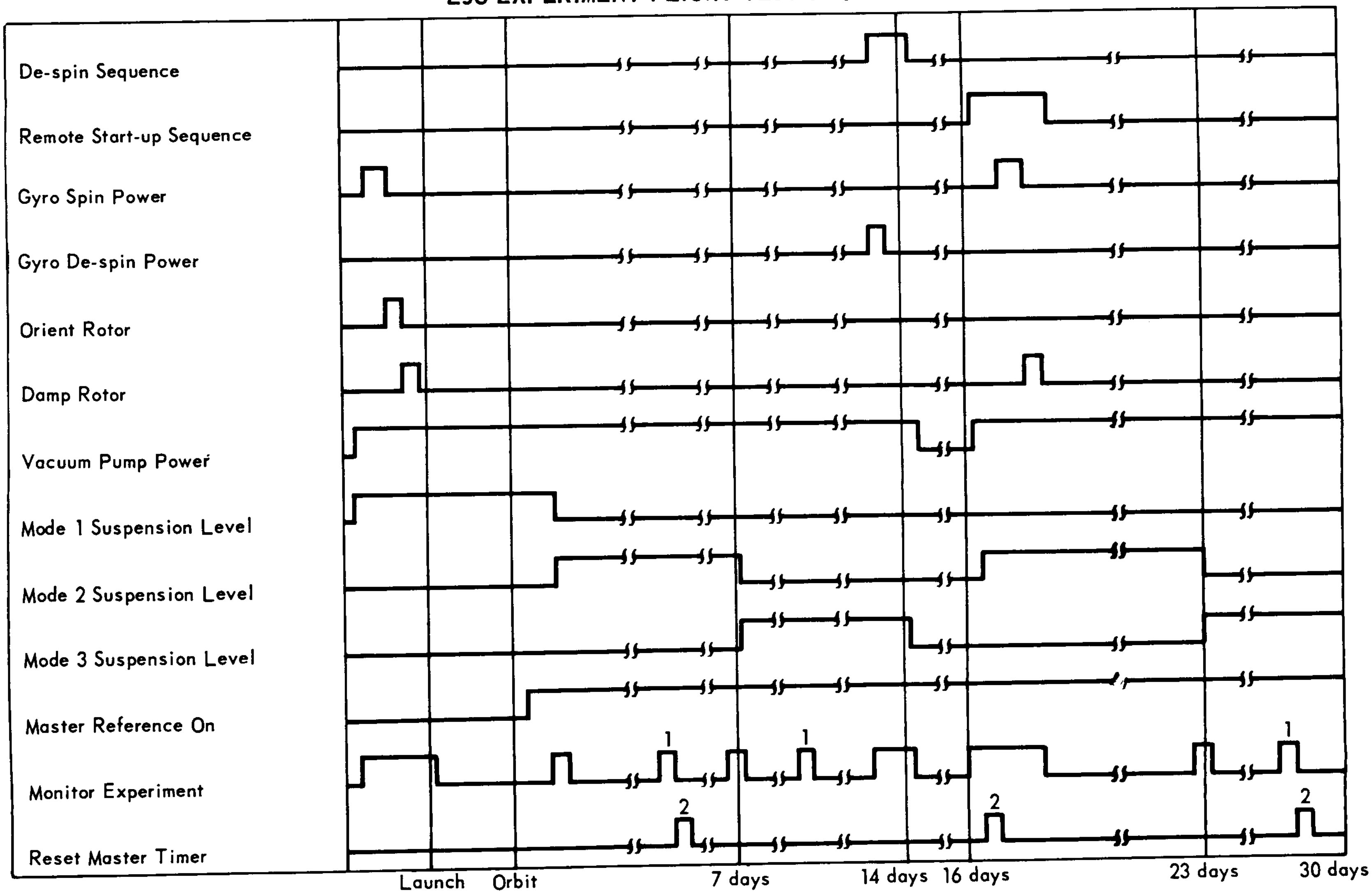
e. Turn master reference on and orient vehicle.

f. Change rotor suspension to mode 2 level.

g. Monitor experiment throughout steps (e) and (f).

h. Monitor experiment for five seconds at ninety minute intervals for one week.

ESG EXPERIMENT FLIGHT TEST SEQUENCE



1 All experiment parameters monitored at 90 minute intervals for one week.

2 Reset at 8 hour intervals through experiment life.

3 Further test cycles dependant upon vehicle life.

FIGURE 2.1-4

i. Change rotor suspension to mode 3 level.

j. Monitor experiment throughout step (i).

k. Repeat step (h).

l. De-spin rotor.

m. Remove rotor suspension.

n. Remove all power to the ESG and its electronics.

o. Monitor experiment throughout steps (l) through (n).

p. Start vacuum pump.

q. Suspend rotor using mode 2 suspension.

r. Spin rotor to full speed and damp nutations.

s. Monitor experiment throughout steps (p) through (r).

t. Repeat step (h).

u. Repeat steps (i) and (j).

v. Repeat steps (h).

Data obtained from this experiment will be processed into a format in which all parameters are correlated to the time reference. The time reference signals will be transformed from "time-since-last-reset" into "real time" signals using the record of "real time/reset time".

Based on the satisfactory conclusion of these tests, further tests on the ESG would appear to be limited to functional tests on the gimballed configuration. In addition, the ESG would appear to have application in other experimental programs as a long term, high accuracy master attitude reference as discussed in Appendix A.

REFERENCES

(1) Welch, J. D., "An Advanced Optical-Inertial Navigation System", AIAA Paper No. 63-215.

(2) Lange, Benjamin, "The Drag-Free Satellite", AIAA Journal, Vol. 2, No. 9, September 1964, pp 1590-1606.

(3) "Orbital Tests for Electrically Suspended Instruments", Honeywell Document No. 4G-D-265, 23 October 1964.

(4) Johnson, Ronald E., "Electrically Suspended Gyros for Space Applications", AIAA Paper No. 63-315.

2.2 Low-G Accelerometer Performance Test

2.2.1 Objectives - The major objective of this test is to measure the bias error or zero offset, scale factor and threshold of an accelerometer designed for operation in the 10^{-4} to 10^{-10} g range.

2.2.2 Background - There are potential applications for accelerometers capable of measuring accelerations as low as 10^{-10} g or having a threshold as low as 10^{-12} g. Accelerometers in this g range can profitably be used for accurate measurement of attitude and/or altitude by determining the magnitude and direction of the gravity gradient vector, and for determining the thrust output of low level thrust motors. Adequate performance evaluation tests cannot be performed on earth for these low range accelerometers. Testing on earth is limited by the effects of cross axis forces due to the one-g earth field, local and seismic induced vibrational noise, and the ability to create accurate acceleration force as low as 10^{-10}g. Orbital test will eliminate such constraints as these, which now place a 10^{-6} g limit on acceleration measurements. At present, two to three orders of magnitude improvement to the 10^{-9} to 10^{-8} range seems attainable by orbital tests. Location uncertainties, tolerances on vehicle angular motion, orbit orientation and prediction uncertainties, etc. introduce errors in this range which place a bound on the measured acceleration. The test method selected for this experiment attempts to minimize the effects of such error sources.

The University of New Mexico and Dynamics Research Corporation of Boston are currently exploring the problem of testing low-g accelerometers. Results of these studies are not yet available, but since they are concentrated in the area of the proposed orbital test, they should be reviewed and incorporated into the test method where applicable.

ARMA, Autonetics, Bell Aerospace, Litton, Nortronics among others are developing accelerometers capable of measuring acceleration in the 10^{-4} to 10^{-10} g range.

LOW-G ACCELEROMETER EXPERIMENT FUNCTIONAL BLOCK DIAGRAM

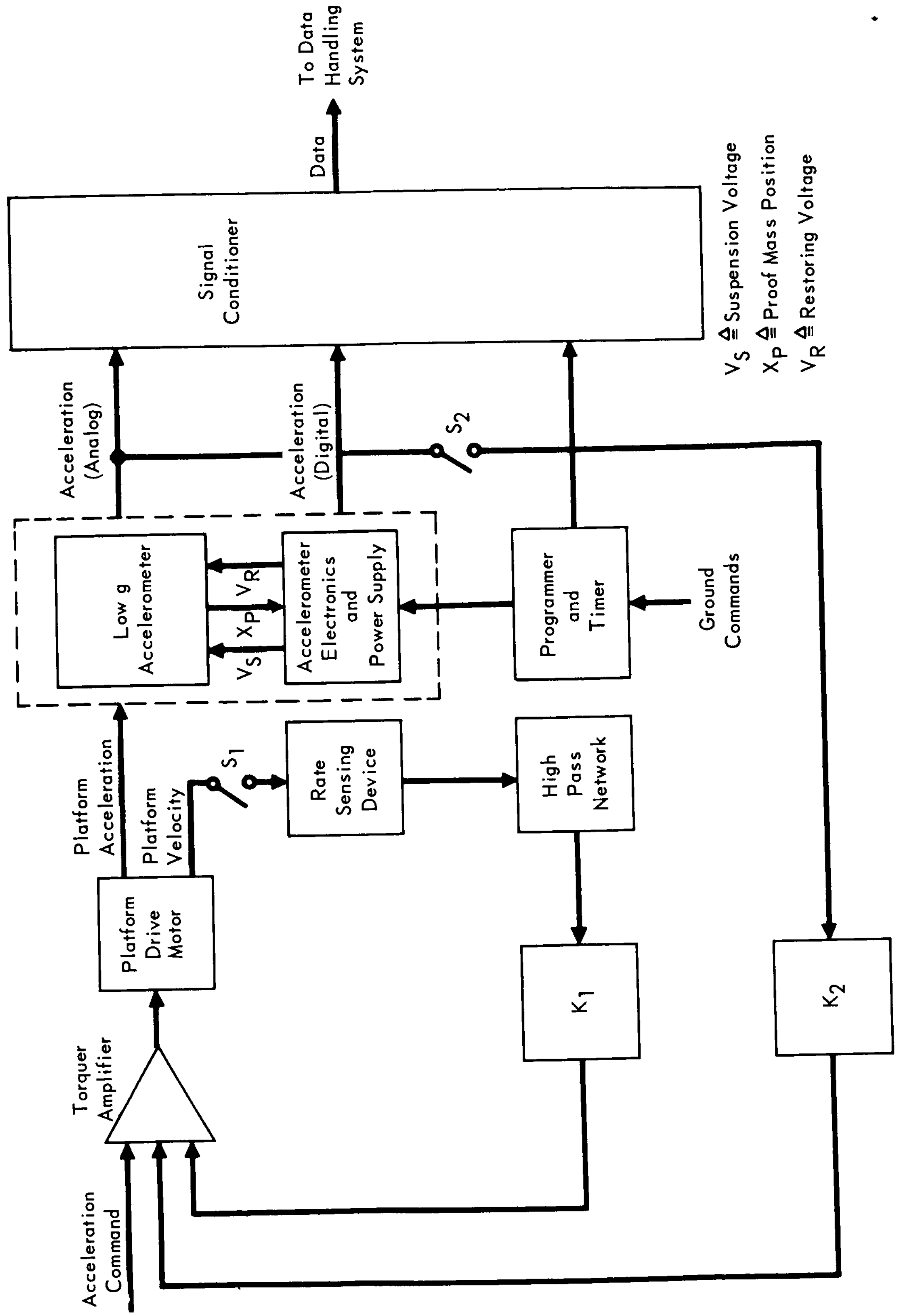


FIGURE 2.2-1

The following description is based upon use of Bell Aerospace Miniature Electrostatic Accelerometer (MESA). The description is applicable to other accelerometers in this acceleration range with minor modifications.

2.2.3 Functional Description - The experimental package will consist of a low-g accelerometer, an accelerometer electronics package, a signal conditioning unit, a programmer and time reference package, and a single axis platform or rotary table with its torquing amplifier. The accelerometer will be mounted on the platform. Two separate procedures are involved in meeting the objectives of the experiment. For the bias error, or zero offset test, the accelerometer sensitive axis will be oriented perpendicular to both the velocity vector and the orbit plane. Accelerometer output will be measured over one complete orbit and then rotated 180° around the proof mass with measurements taken for one additional complete orbit. The two measurements will be summed and the resultant will be attributed to zero offset. The scale factor test will utilize a platform to torque the accelerometer until it is deflected a known distance. Correlation of measured data will then be used to determine scale factor error. A functional block diagram of the experiment is shown in Figure 2.2-1. A description of the experiment equipment, test method and major error sources is contained in the following paragraphs.

Experiment Equipment - A view of the accelerometer components is shown in Figure 2.2-2. The accelerometer proof mass consists of a hollow cylindrical float which has a flange around the center. The cylinder is supported by eight electrodes arranged to act against the inner surface of the float. Each electrode is in series with a signal generator and a tuning inductor resulting in a voltage across the electrode-float gap which is a function of the float position. The float position determines the capacity in a series tuned circuit and, therefore, the operating point on the series resonance characteristic curve. As a result of

ACCELEROMETER COMPONENTS

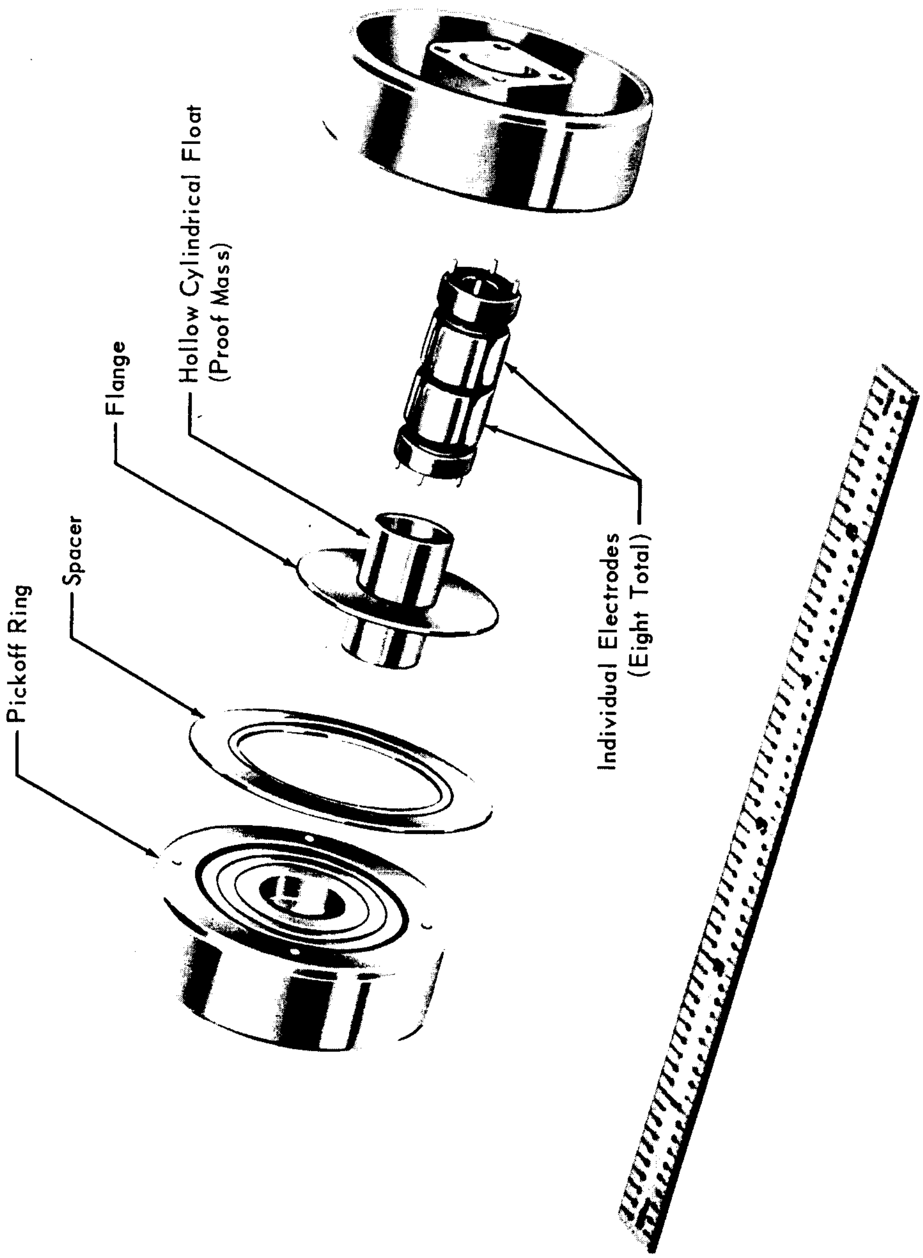


FIGURE 2.2-2

this situation, attractive forces develop between the float and the electrodes, so that the float is suspended in all directions except along the cylindrical axis. The suspension system, therefore, centers the float under angular and translational accelerations. The suspension sweep oscillator and amplifiers provide automatic suspension of the float without voltage breakdown or arcing.

Detection of the float movement is accomplished by sensing the change in capacity between the pickoff rings and the float flange. When used in a balanced capacitive bridge circuit excited by a carrier frequency, displacement of the float results in a signal output from the bridge. This output is amplified and demodulated by a phase sensitive detector to produce a d-c voltage whose polarity is a function of the direction of the float movement and whose amplitude is a measure of the relative displacement of the accelerometer null position. This voltage, when applied to the trigger circuit, results in the generation of a d-c pulse which restrains the float.

The forcing of the float along the sensitive (cylindrical) axis is accomplished using rings which are capacitively coupled to the float flange. Voltage pulses are applied to these rings to restrain the float whenever a force is applied along the sensitive cylindrical axis. The pulses are generated by a pulse core which controls the amplitude and width of the pulse. The system output information is a signal whose frequency is proportional to acceleration (each pulse is equivalent to an incremental velocity change).

A block diagram of the accelerometer electronics is shown in Figure 2.2-3. The suspension electronics consists of a crystal controlled sweep oscillator. The sweep feature of the oscillator permits the lifting of the float from rest as well as maintaining suspension.

ACCELEROMETER ELECTRONICS

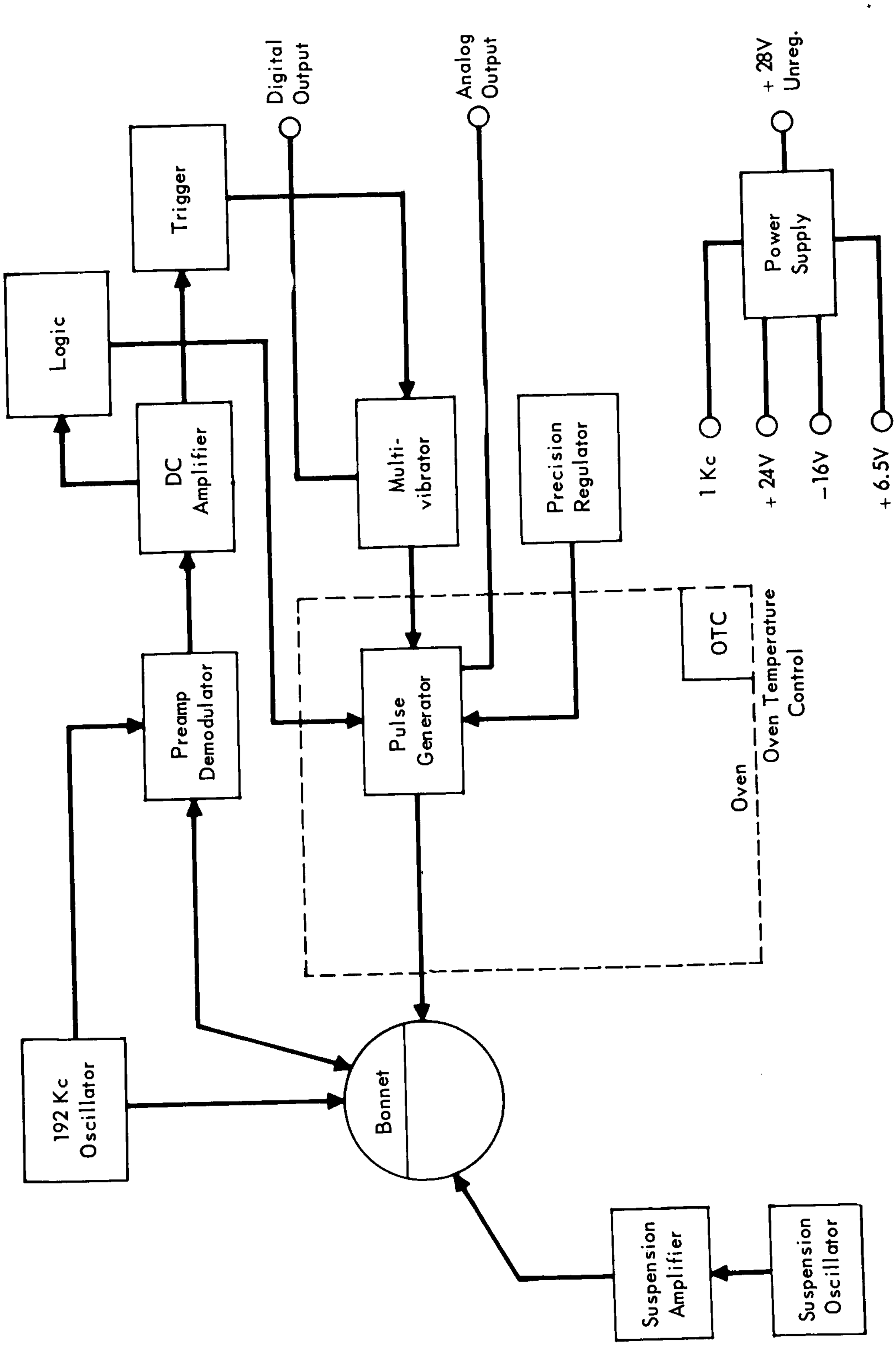


FIGURE 2.2-3

The pickoff system consists of an excitation oscillator, a preamplifier demodulator and a d-c amplifier. The oscillator provides the accelerometer bridge excitation and a reference signal for the phase sensitive demodulator of the pickoff signal. The preamplifier-demodulator output is amplified and supplied to the pulse generation electronics. The pulse generation electronics provides a pulse restoring force to the accelerometer. It consists of a trigger logic-circuit, a precision voltage regulator and a pulse generator. When the input to the pulse generation electronics reaches a predetermined level (trigger level), a pulse is generated. The pulse width and amplitude are closely controlled since this is the measure of incremental velocity.

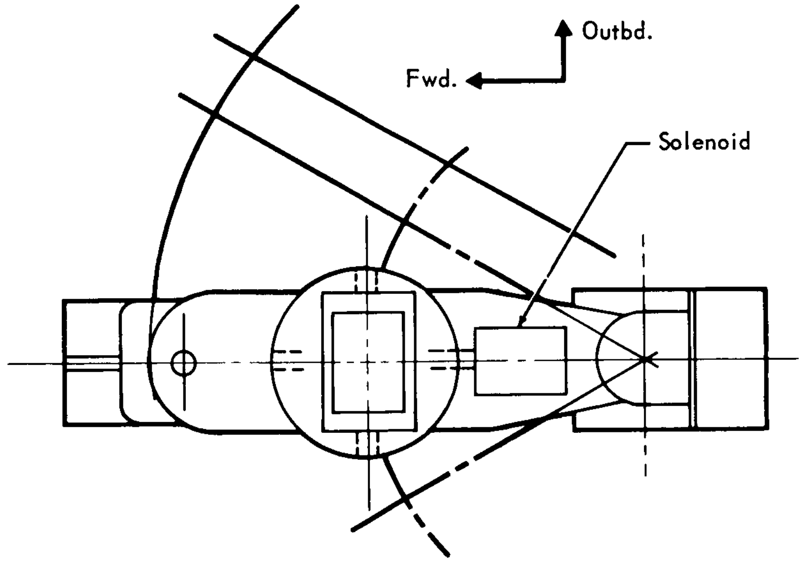
The accelerometer installation is shown in Figure 2.2-4. An inductive type transducer is used to accurately measure the distance through which the accelerometer has been translated. No slip rings are required since the accelerometer is rotated less than 360 degrees. Acceleration and vibration induced noise are minimized by bearing design for this application.

A motor provides the capability of indexing the accelerometer-plus sensitive axis in any of 4 position displaced multiples of 90 degrees from the center line. A d-c torquer provides power for displacing the accelerometer by rotation about the bearing axis. The torque amplifier consists of a preamplifier and a power amplifier, which supplies current for the d-c torquer.

The time reference system contains a clock which has the capability of being started, reset and held as commanded. Digital time readout of the clock is required. The programmer provides the logic and sequencing of the experiment.

The spacecraft equipment requirements consist of a command receiver, a signal processor and encoder package, and a telemetry transmitter. A tape recorder may be required for vehicles with limited data transmission capabilities for this experiment.

LOW-G ACCELEROMETER INSTALLATION



Data: (Approx.)
Weight – 6.2 lb.
Volume – 1 cu.ft.
Outline Dims. – 8.4" x 12.8" x 16"
(Includes Acc. Travel)

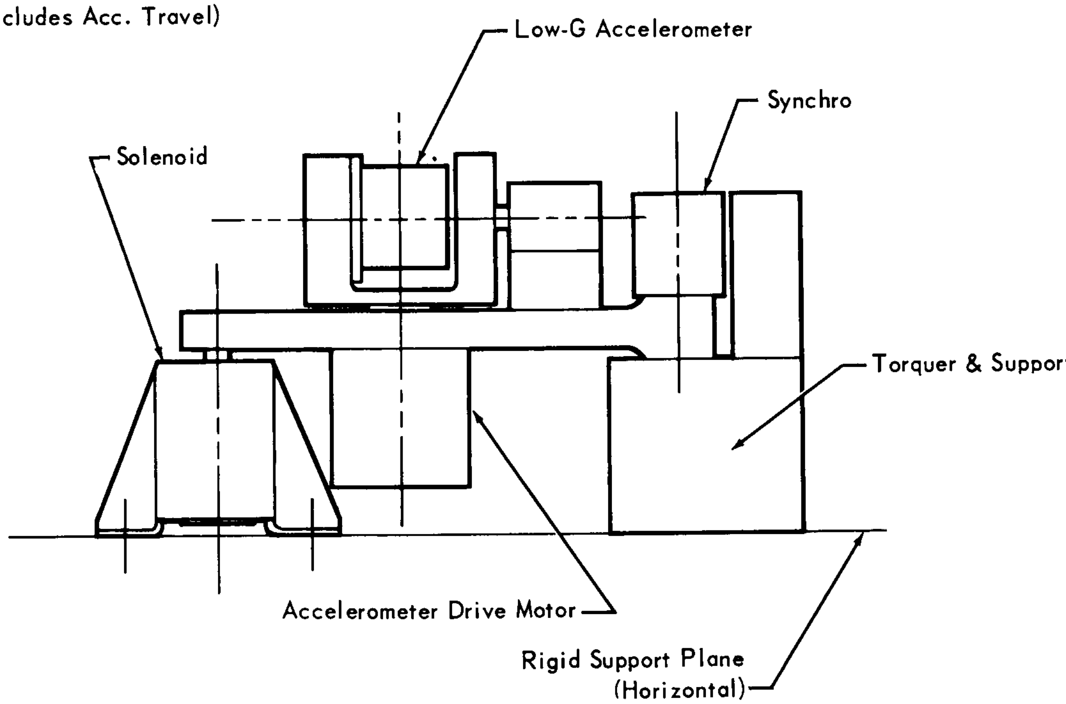


FIGURE 2.2–4

Test Method - The vehicle is controlled to an earth local vertical orbit plane alignment in which body axis angular rates must be nearly zero. A circular orbit is required to prevent variations in aerodynamic drag.

Zero offset or bias error is obtained by the following procedure: The accelerometer sensitive axis is positioned perpendicular to both the velocity vector and the orbit plane with the proof mass located near the vehicle center of mass. Prior to start of the test, the time reference, the velocity and distance counters are reset and held at zero. Upon command the experiment is started. After one orbit, the outputs from velocity counter, distance counter and time reference are recorded or transmitted. This procedure is repeated with the accelerometer sensitive axis reversed by rotating the accelerometer 180 degrees around the accelerometer proof mass in order to eliminate the effects of vehicle gravitational attraction forces. Acceleration is calculated from the equation

$$v = at \quad \text{or} \quad d = 1/2at^2$$

Because of the selected orientation of the accelerometer sensitive axis, the bias error is simply the algebraic sum of the two calculated accelerations. Vehicle acceleration in this axis is the algebraic difference of the two calculated accelerations. Since this procedure assumes the vehicle acceleration to be constant for each orbit, the procedure should be repeated several times and the data correlated to obtain the best estimate of bias error.

Scale factor is obtained by the following procedure. The output of the accelerometer is fed into the platform torquer to create a closed loop acceleration system. Prior to the start of the test the platform is rotated to the zero position by feeding a zero position signal into the torquer. The time reference counter, velocity counter and distance counter are reset to zero. Upon command at $t = 0$, the zero position signal is switched out and platform position is controlled by a signal comprising the sum of accelerometer output and a programmed command. This will result in the accelerometer output being driven to a value equal to the programmed acceleration command. When the accelerometer has been rotated through the selected angle corresponding to a position change of either 3 inches or 0.3 inch, the velocity counter, time counter and distance counter readings will be recorded or transmitted to a ground receiving station. This procedure will be repeated for different levels and directions of acceleration. Scale factor will be computed by correlating the measured data. The time required to traverse a distance of 3 inches or 0.3 inch for constant acceleration levels between 10^{-4} g and 10^{-10} g are shown in Table 2.2-1.

An alternate scheme to obtain a closed loop acceleration system for the scale factor is being considered. Platform angular velocity will be sensed by a very accurate rate gyro or some precision rate sensing device and compensated by a high

TABLE 2.2-1

TRAVERSE TIME AS A FUNCTION OF ACCELERATION

ACCELERATIONS (g)	TIME TO TRAVERSE GIVEN DISTANCE (SECONDS)	
	3 INCHES	0.3 INCH
10^{-4}	12.5	3.94
10^{-5}	39.4	12.5
10^{-6}	125	39.4
10^{-7}	394	125
10^{-8}	1250	394
10^{-9}	3940	1250
10^{-10}	12500	3940

pass network to provide a feedback signal to the platform torquer proportional to acceleration in the frequency range of interest. This signal would supplement or replace the accelerometer feedback to the torquer, depending on the results of lab tests. The advantage would be higher accuracy in maintaining a constant platform acceleration. This could be verified by ground tests and analysis. Both feedback loops are shown in Figure 2.2-1, with switching S_1 and S_2 provided to permit switching either loop out.

Vibration and accelerometer threshold data will be obtained by feeding proof mass position into the platform torque amplifier. By measuring the proof mass position, qualitative data can be obtained to indicate vibration and threshold levels. Low frequency steps in the proof mass position would indicate threshold, whereas high frequency signals would indicate vibration.

Additional test information may be obtained by changing the suspension force level and scaling the accelerometer to measure a lower g range. Scaling is readily done by programming the suspension system and the pulse width of the restoring torque.

Major Error Sources - The extremely low level acceleration measuring capability of the device results in output measurements highly sensitive to contamination by equally low level error producing sources. It is not currently within the state-of-the-art to control vehicle motion and environmental factors accurately enough to completely eliminate such sources of error. Major error sources include: (1) uncertainties in center of mass location, (2) motion of masses within the satellite, (3) thermal noise causing random motion of accelerometer proof mass, (4) vehicle angular accelerations, (5) gravitational anomalies (affecting repeatability of test), (6) orbit uncertainties, (7) drag, (8) solar forces, (9) micrometeorite showers, (10) accelerometer and platform alignment errors, (11) cross-coupling between accel-

TABLE 2.2-2

LOW-G ACCELEROMETER
EXPERIMENT PHYSICAL PARAMETERS

ITEM	SIZE (INCHES)	VOLUME $FT.^3$	WEIGHT (LBS.)	OPERATING POWER (WATTS)
Accelerometer	2.75 dia. x 2.4	.01	1.0	5
Accel. Electronics and Power Supply	7.7 x 5.4 x 2.5	.06	12.5	5
Platform	8.4 x 12.8 x 16.0	1.0	16.2	4
Torquing Amplifier	2 x 3 x 3	.01	1.5	8
Programmer and Timer	5 x 5 x 3.5	.05	3.5	6
Signal Processor and Encoder	4 x 4 x 2	.02	1.0	7
Total	2075 cu.in.	1.15	35.7	35

erometer axes, (12) platform transients due to power supply fluctuations, and (13) resolution.

Of these error sources, accelerometer location with respect to the vehicle c.m. and vehicle angular rates will be the most serious. For example, a location uncertainty, δr, and vehicle rate, ω, causes a component of acceleration equal to $\delta r \omega^2$ which is independent of the direction of vehicle rotation and cannot be averaged.

2.2.4 Experiment Physical Parameters - The size, weight, and power of the experiment packages are contained in Table 2.2-2.

2.2.5 Data Parameters - Table 2.2-3 lists the important experimental parameters to be measured during the test. Signal form, parameter range, accuracy and sample rate are listed along with pertinent remarks. The experimental device data parameters are shown in the first five items. The accuracies called out in the range of parameter column were calculated for acceleration levels of 10^{-10} g.

TABLE 2.2-3

LOW-G ACCELEROMETER EXPERIMENT DATA PARAMETERS

DATA POINT	SIGNAL FORMAT	PARAMETER RANGE	SAMPLE RATE (BANDWIDTH)	ACCURACY	RECORD	ESSENTIAL	DESIRABLE	CONTINUOUS
Proof Mass Position	Analog Voltage		3 samples/sec. 1500 samples/sec.	±3%	X	X		
Acceleration	Analog	$+10^{-6}$ to -10^{-6}g	3 samples/sec.	±3%	X	X		X
Velocity	Digital	$\pm 10^{-3}$ ft./sec. 20 bits	Readout once per test	±1 bit	X	X		X
Distance	Digital	± 2 feet 20 bits	Readout once per test	±1 bit	X	X		X
Time	Digital	0–10,000 sec. 14 bits	Readout once per test	±1 bit	X	X		X
Yaw Position	Digital	± 5° (14 bits)	1 sample/sec.	±1 bit	X	X		X
Roll Position	Analog	±10°	3 samples/sec.	±3%	X	X		X
Pitch Position	Analog	±10°	3 samples/sec.	±3%	X	X		X
Power Input	Analog	20–35 volts	3 samples/sec.	±3%	X		X	X

SIGNAL CONDITIONER FOR DIGITAL DATA

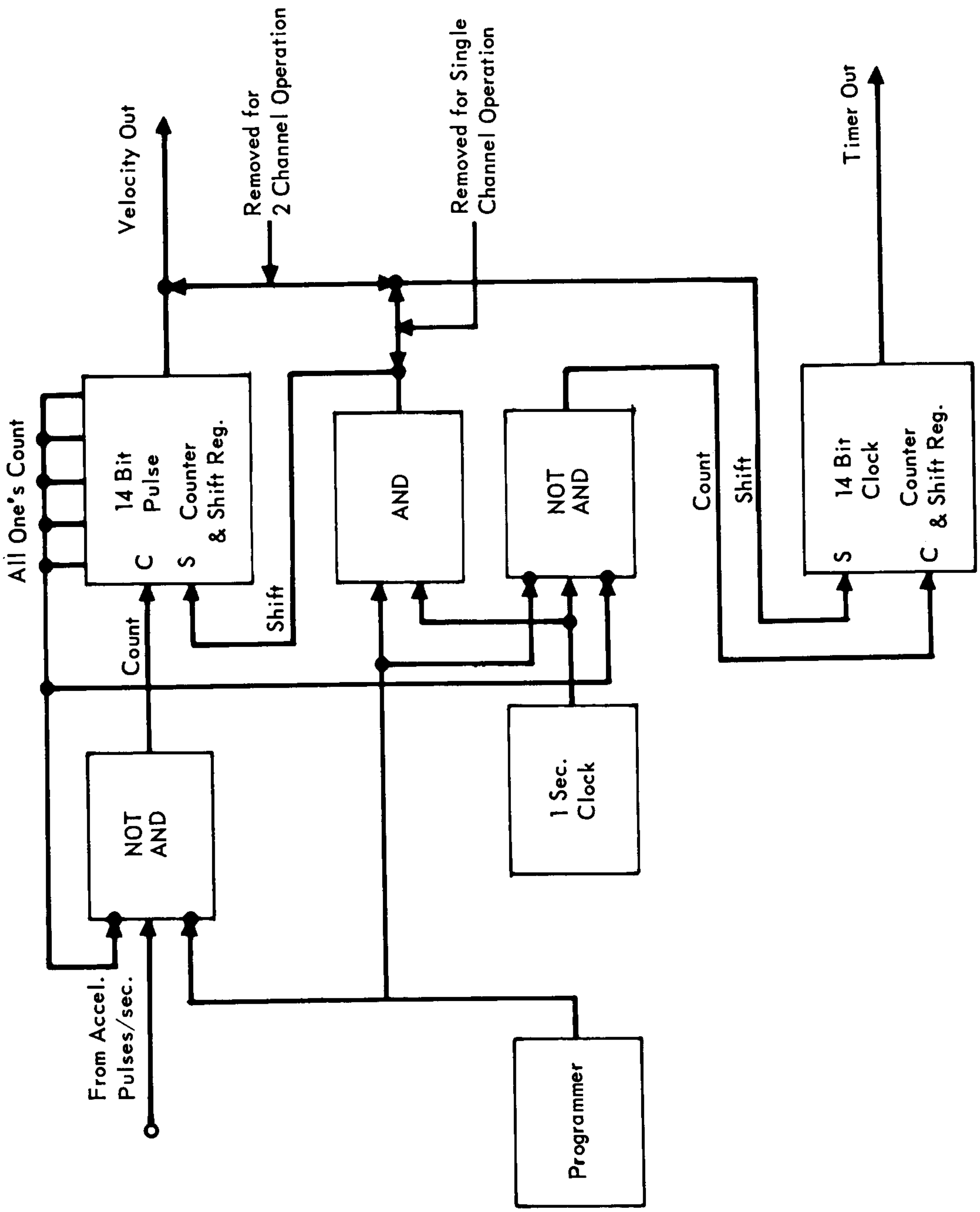


FIGURE 2.2-5

Although realistic appraisal of error source magnitudes and component state-of-the-art considerations indicates an expected accuracy of about 10^{-8} g, the instrumentation will not be limited there. One reason for performing the experiment is to determine the type and magnitude of system errors for later application.

Incremental velocity is determined by counting the number of pulses necessary to restrain the float (proof mass) at some point. The total number of pulses is accumulated in a storage register and later sampled at a nominal rate of one sample per second. Velocity is obtained by counting the total number of pulses accumulated in the storage register. The output signals are compatible with data handling requirements. Acceleration and proof mass position are also available as analog signals. In addition to the device data parameters, yaw, roll and pitch position are recorded at the same rates specified in Table 2.2-3.

2.2.6 Vehicle Orbit and Attitude Requirements - A near circular orbit (eccentricity, $\epsilon \leq .01$) having a minimum orbital altitude of 300 nautical miles is desired for the test. Orbit inclination should be chosen to maximize data transmission capability. Orientation in yaw, roll and pitch must be held to within $\pm 0.5^\circ$ of the chosen local vertical/orbit plane alignment. Body axis angular rates about the yaw axis must be less than .08 milliradians per second. Rates about roll and pitch are not critical except for yaw cross-coupled rates.

2.2.7 Experiment Support and Data Handling Requirements

Experiment Support - The use of analog data signals having a wide dynamic range encounters threshold and resolution problems. When possible, these difficulties are circumvented by using digital signals. Special data conditioning techniques must be employed to match the digital signals to the data handling system. Figure 2.2-5 shows a typical example of the logic elements required for digital signal conditioning. Such a circuit would be employed to handle the velocity data

LOW-G ACCELEROMETER FLIGHT TEST DEVELOPMENT PLAN

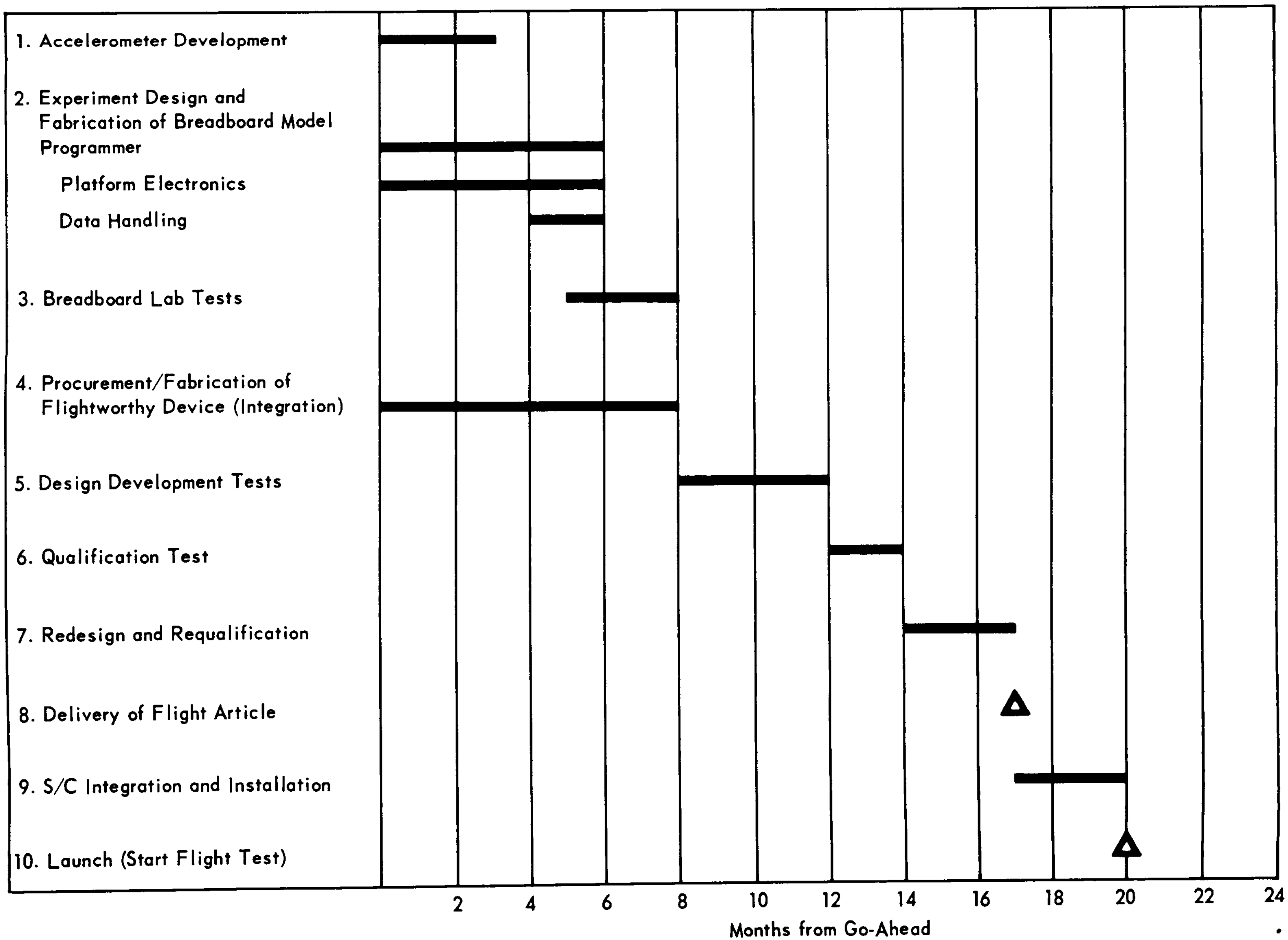


FIGURE 2.2-6

parameter. An additional signal conditioning requirement is to include a circuit to superimpose the end of travel indication on the analog acceleration channel.

Temperature and vibration environmental constraints must be placed on the vehicle. The non-operating temperature limits for the MESA 1C are 30°F to 160°F. During operation the upper limit is reduced to 100°F.

The experimental device and associated equipment require 31 watts. Approximately 300 watt-hours are required for the entire test.

The only experiment ground commands required are to initiate sequence, and to dump data.

Data Handling - The basic data handling requirements are similar to those defined by Appendix B. The analog acceleration data will be recorded for the duration of the test. The data will be sampled at a 3 samples/second rate or greater, except for vibration data, where the requirement for a 1500 samples/second rate exists. This high rate requirement can be met by changing the sample rate for a 2 or 3 second burst of 1500 samples/second for each test. A corresponding increase in the recorder speed may be required.

2.2.8 Experimental Flight Test Plan - The development program is geared to deliver the flight article at the end of the 17th month following go-ahead. Significant milestones in the program along with completion times are shown in the bar graph of Figure 2.2-6.

Two separate tests will be made during flight. At least 5 runs of each test will be made to insure that sufficient data samples are available to obtain meaningful results. The experiment operational sequence for the bias error tests is shown in Figure 2.2-7 and for the scale factor test in Figure 2.2-8.

The test is initiated by ground command "INITIATE SEQUENCE". This energizes the on-board programmer and master reference. The master attitude reference is

LOW-G ACCELEROMETER BIAS ERROR PROCEDURE

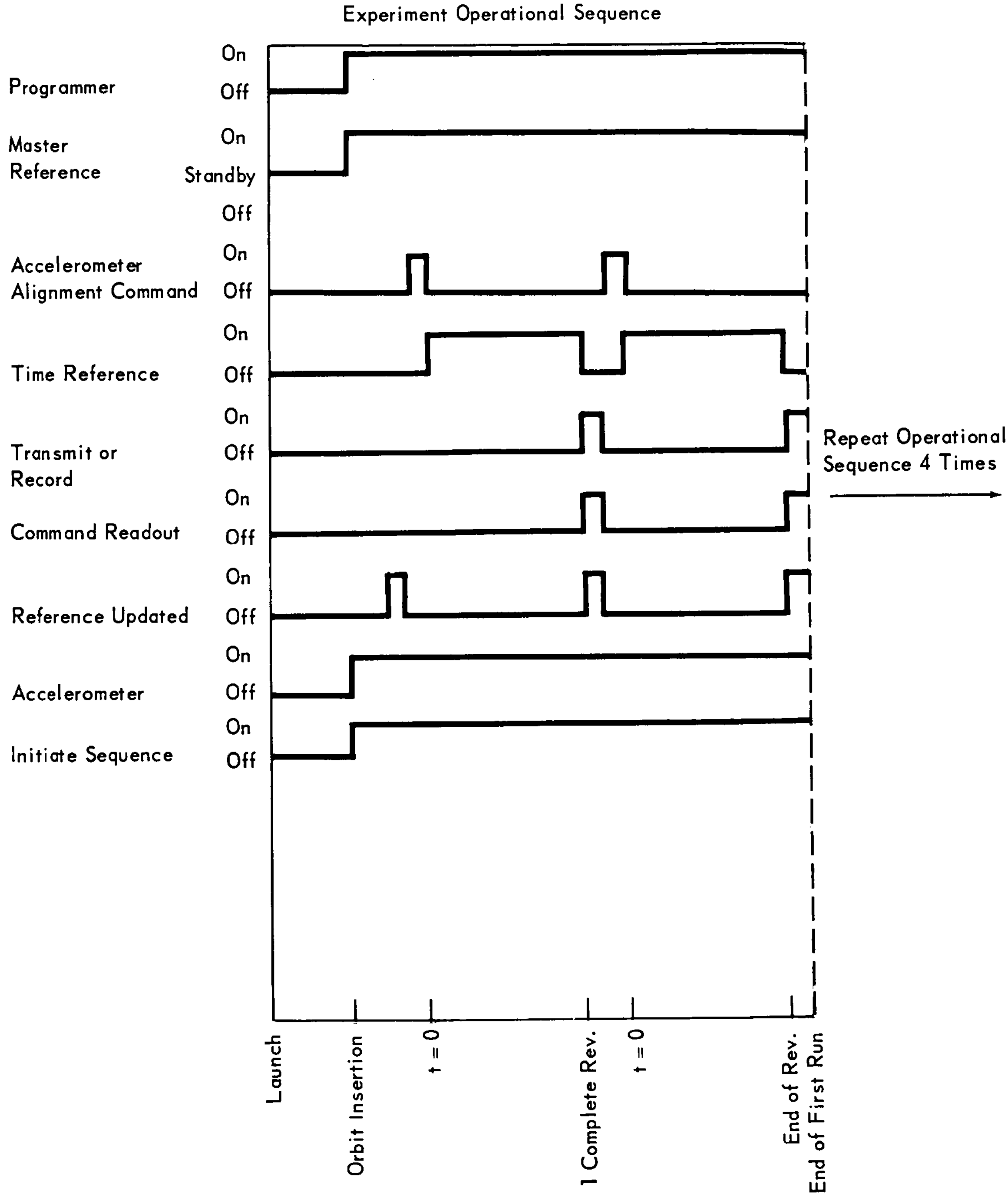


FIGURE 2.2-7

LOW-G ACCELEROMETER SCALE FACTOR PROCEDURE
(Follows Bias Error Test)

Experiment Operational Sequence

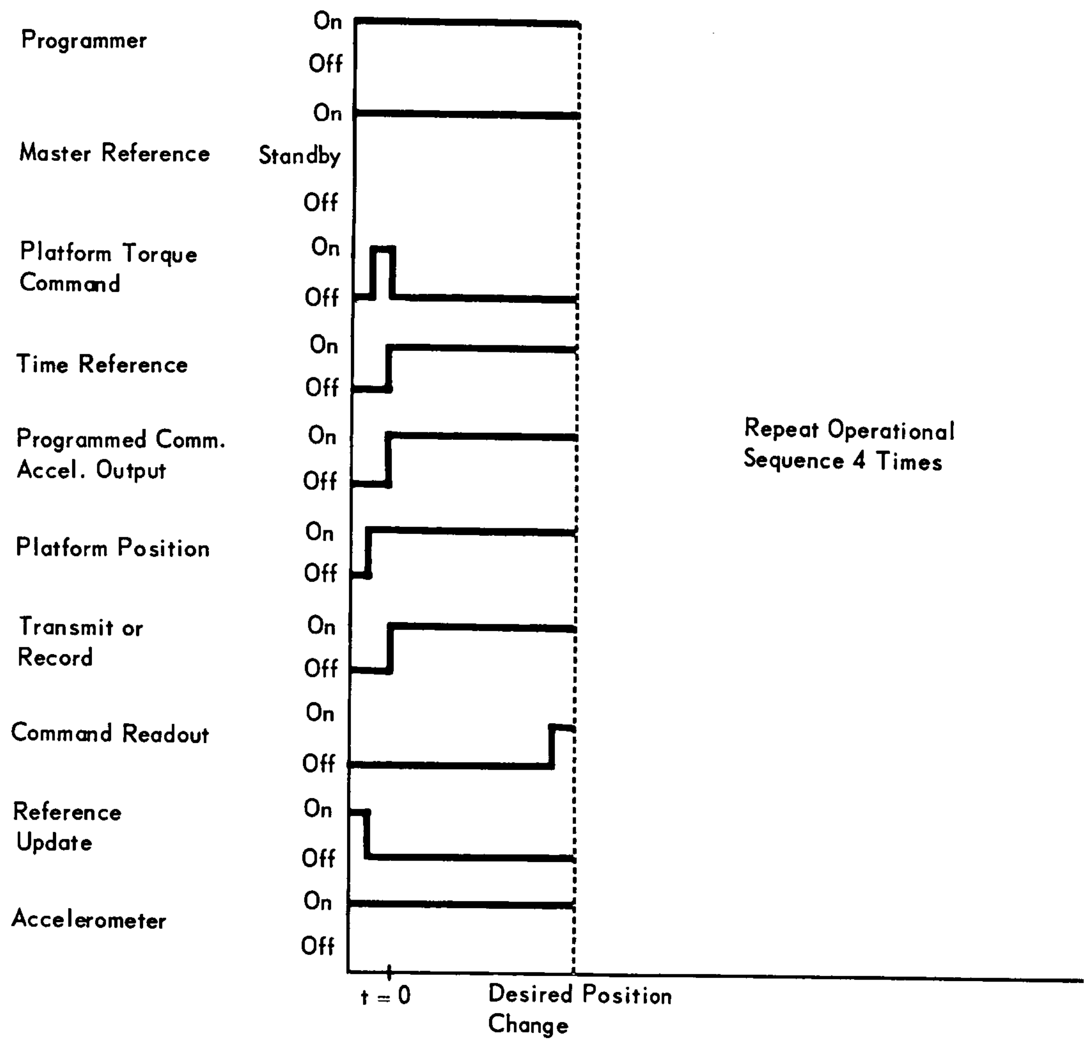


FIGURE 2.2-8

first updated and then used as the reference for automatic alignment of the accelerometer perpendicular to the velocity vector and orbit plane. The time reference counts time until one complete revolution in orbit has occurred. An accelerometer alignment command to rotate the sensitive axis 180° is then given and time is counted for another complete revolution. This sequnce is repeated until five complete test runs have been made.

The scale factor test procedure is initiated by ground command. The platform, on which the accelerometer is mounted, is torqued until it reaches a predetermined zero position. A plus acceleration output is commanded to feed the accelerometer output into the platform torquer amplifier. Time reference, position meter, velocity meter, and recorders are automatically switched in along with an acceleration command signal to the torquer amplifier. The test lasts until the platform angular position indicates that the accelerometer has traveled through the selected distance, at which time a signal causes the data sensors to stop operating. This sequence will be repeated until five runs have been made. A ground command to dump data will be given at an appropriate point in orbit.

2.3 Gravity Gradient Sensor

2.3.1 Objective - The suggested gravity gradient sensor orbital tests are intended to:

a. Demonstrate the accuracy of the device in a space environment.

b. Measure error producing sources to facilitate ground testing.

c. Obtain design data for this and other approaches to gravity gradient sensing of the local vertical.

The gravity gradient sensor, if demonstrated to achieve its potential accuracy, would fill a need as the navigation attitude reference for precise navigation systems.

2.3.2 Background - Measurement of certain basic physical properties of the earth's gravitational field will uniquely determine the direction of the earth local vertical. The gravitational field of an approximately spherical mass such as the earth decreases with altitude because of the inverse square law, and changes direction with horizontal displacement because of curvature of the equipotential surfaces. Changes in gravitation around a circle with origin at reference point O are shown in Figure 2.3-1a. Gravitation, $\overline{G}$, at points R, near reference point O, differ from gravitation, $\overline{G}_o$, at O by difference $\overline{g} = \overline{G} - \overline{G}_o$ shown in Figure 2.3-1b. The tangential component, g_θ, of this change at points on a circle of radius r about O is shown in Figure 2.3-1c. It is observed that this component varies sinusoidally with angular position θ around the circle, as indicated in Figure 2.3-1d, and completes two cycles per revolution. The direction of the vertical corresponds to a value of θ for which the sinusoid goes through zero.

A device that utilizes these gravitational effects can provide a passive means of determining the local vertical. The potential accuracy of this method

GRAVITATIONAL FIELD NEAR A REFERENCE POINT OVER EARTH

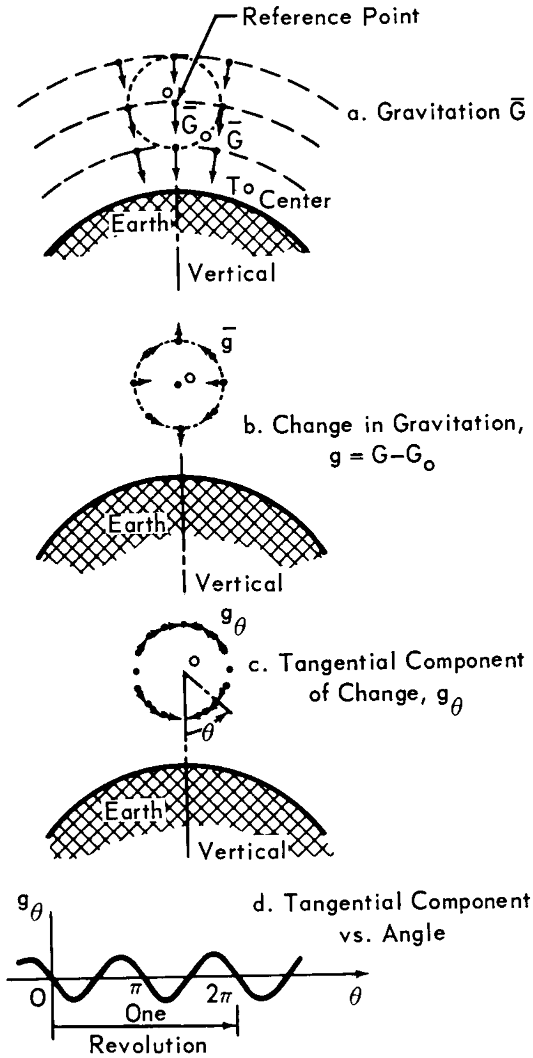


FIGURE 2.3-1

.due to uncertainties in the direction of the vector gradient is on the order of 2 seconds of arc. Reference (1) describes the principles of operation and the essential components of a device to determine local vertical by utilizing the gravity gradient effect.

This approach uses a sensitive (low-g threshold) accelerometer, rotated at a constant angular rate, to sense the sinusoidal variation in the tangential component of the gravity gradient acceleration, g_θ. Utilization of the accelerometer in this manner eliminates the adverse effect of accelerometer bias and scale factor errors. In this method the sensed quantity, g_θ, is a sinusoid having a phase error proportional to the error in the vertical.

At present a device utilizing the approach is not fully developed. A sufficiently sensitive low g accelerometer (10^{-13} g threshold) is required to achieve measurement accuracy approximating the uncertainty in the knowledge of the direction of the gravity gradient. A number of low g accelerometers are currently being developed, which could be used in this experiment. The fabrication of the remaining components of the complete gravity gradient sensor system is a matter of technique and will not require any significant advances in the state-of-the-art. Certain basic ground tests of the wheel suspension, pick-offs, methods of information processing, etc., are currently being formulated at Litton Systems, Inc/Guidance and Control Division to further develop the approach.

Space testing of the gravity gradient sensor is required because the one-g gravitation acceleration existing on the ground and the seismic noise level of the earth are many orders of magnitude above the acceleration levels to be measured in orbit. This approach has the capability of separating the error producing sources and thereby provides data application to other approaches.

GRAVITY GRADIENT SENSOR EXPERIMENT FUNCTIONAL BLOCK DIAGRAM

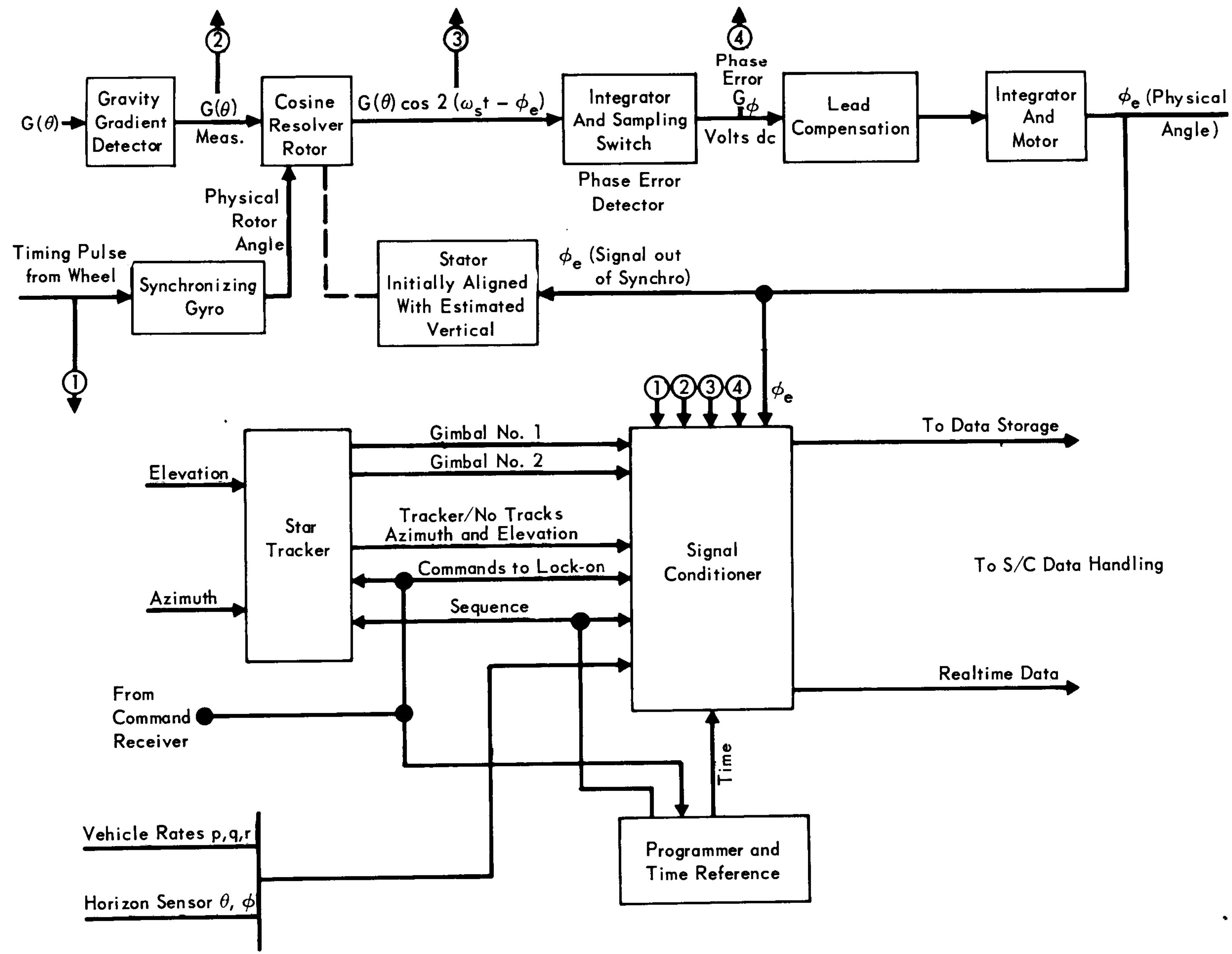


FIGURE 2.3-2

2.3.3 Functional Description - Figure 2.3-2 is a functional block diagram of the experiment. It includes the experiment equipment and on-board sensing devices required for vehicle orientation and establishing a reference (local vertical). This section will describe these system components and their operation.

The goal of this experiment is to measure the local vertical to an accuracy of 1.0 milliradian. A single plane mechanization of the gravity gradient sensor is used to decrease complexity, while demonstrating the feasibility of the approach. A low-g accelerometer is mounted on the rim of a rotating wheel with the sensitive axis mounted perpendicularly to the radius of the wheel (see Figure 2.3-3). The accelerometer output, $-g_\theta$, continuously resolved through an angle $2\theta_e$ measured from a reference axis, Z_e, is also shown. The resolver rotor moves with the spinning wheel, whereas the stator is initially positioned along the assumed direction of the vertical and defines the reference axis, Z_e. The angle

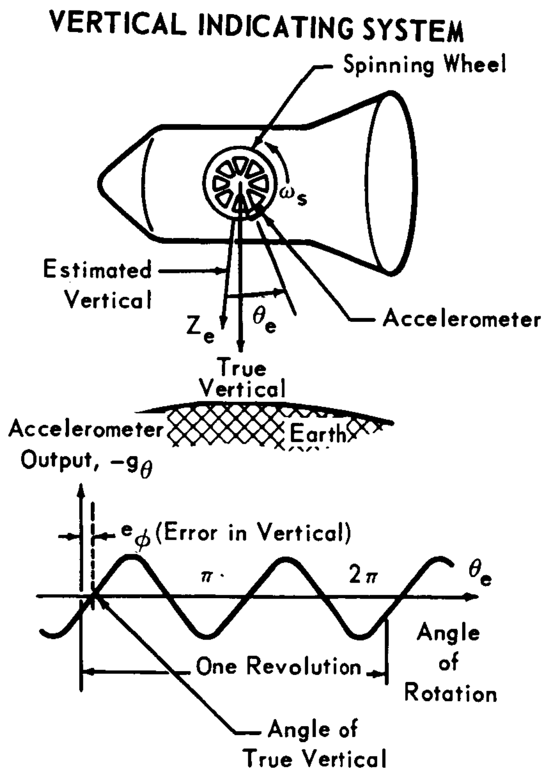


FIGURE 2.3-3

of the stator with respect to the vehicle body reference axis is accurately measured by a digital shaft encoder.

The resolver is coupled in the following manner: a light source is located on the sensitive element wheel. It is masked out except for two small sectors separated by 180° of arc. A photocell is arranged to detect the light pulses produced by rotation of the wheel at 2 pulses per revolution. These pulses are compared time-wise to the synchro pulse produced by the driver of the resolver. The operating point picked will correspond to a fixed pre-set phase delay between the resolver and the accelerometer output. The detected pulses are used to monitor the rotor shaft of the resolver (S_r) and generate a sampled data indication of angular error to the rotor shaft motor, θ_e, as shown in Figure 2.3-4. These timing pulses are also recorded on the time reference which is telemetered back

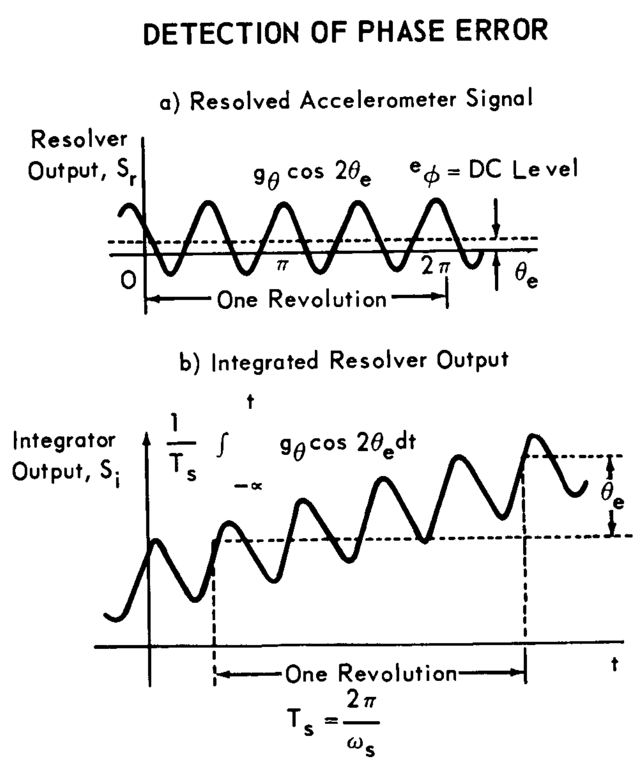


FIGURE 2.3-4

to earth. Because of the constant angular rate of the wheel between pulses, the pulses are adequate to continuously describe the angle of the wheel.

Determination of the phase of the signal sensed by the accelerometer is achieved by a phase-tracking loop as indicated schematically in Figure 2.3-5. It will be observed that this tracking is achieved by comparing the indicated phases with the actual phase in a phase-error detector, the output of which drives the indicated phase until this error, e_ϕ, is nulled.

To detect the error in phase, a reference sinusoidal signal is generated at exactly twice the disk frequency but lags the incoming signal in phase by approximately 90° as indicated in Figure 2.3-6. The frequency of the reference signal is accurately controlled by using the disk itself in the generation of this sine wave. The phase of the reference signal is controlled by the indicated phase. The incoming signal from the accelerometer is then modulated by this reference signal as shown. When the error in phase is zero, the reference signal is exactly

DIAGRAM OF PHASE-TRACKING SYSTEM

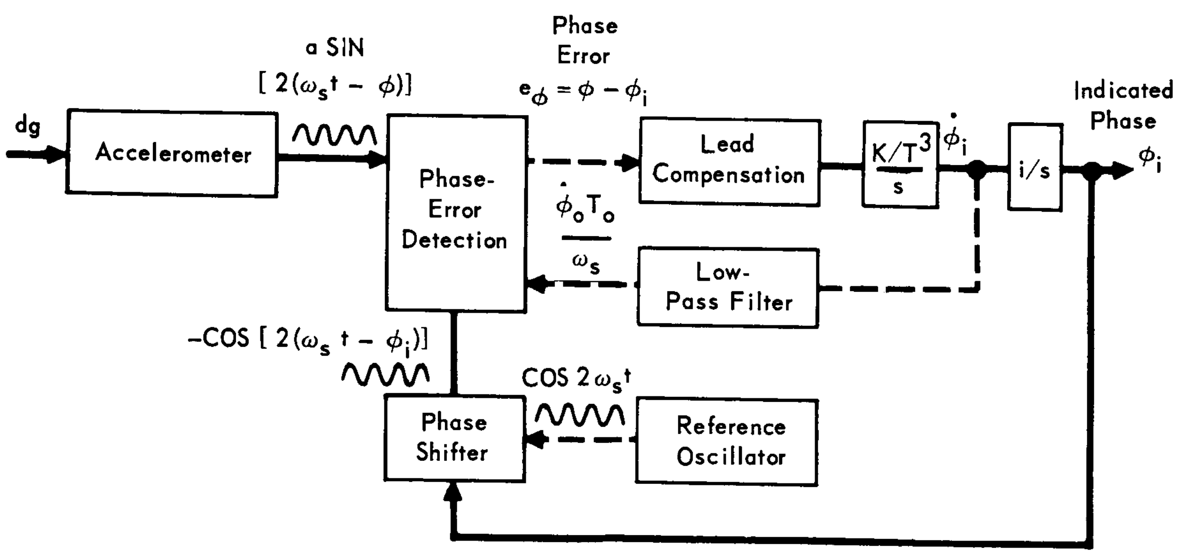


FIGURE 2.3-5

PHASE ERROR DETECTOR

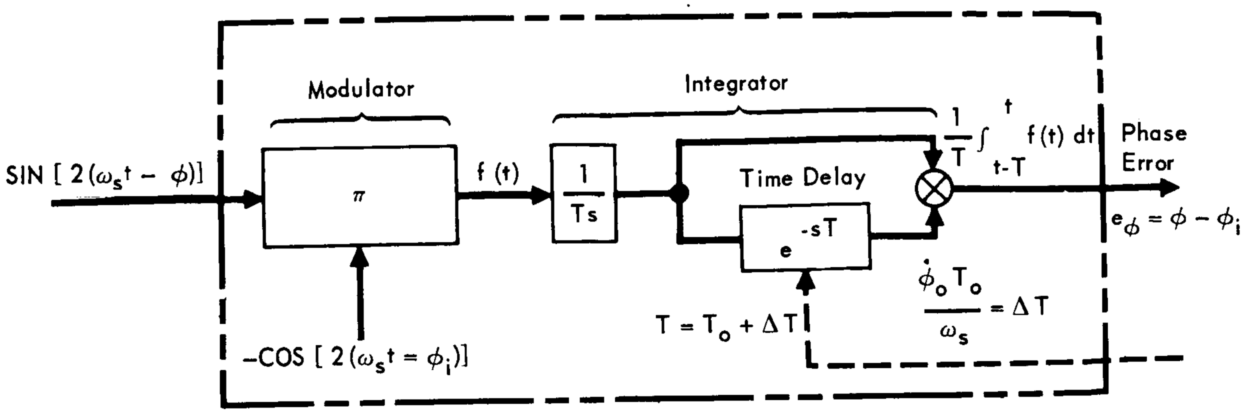


FIGURE 2.3–6

90^o out of phase with the incoming signal and the product has a zero d-c component. If it is not zero, the d-c component of the product is an indication of this error in phase. The d-c component of the modulated signal is measured by integrating over an integral number of cycles of the signal frequency.

The use of the error signal e_ϕ in a closed loop vertical tracking system is accomplished by including two integrations in the loop to avoid steady-state error due to the orbital angular velocity of the true vertical. The effect of the phase error detector is to introduce the transfer function $(1 - e^{-Ts})/T_s$, which will require lead compensation for stability.

The angle of the vertical with respect to a reference axis in the vehicle is obtained to within an accuracy of one milliradian by using a digital shaft encoder to measure the angle θ_i between the stator and the reference axis when the phase error has been driven to zero. Since the master reference provides the angle between the reference axis and the vertical, θ_T, the error, θ_E, between the indicated vertical, θ_i, and the true vertical, θ_T, is $\theta_E = \theta_T - \theta_i$.

<u>Experiment Equipment</u> - The gravity gradient sensor will consist of an electrostatically suspended, freely spinning inertia wheel with a low-g accelerometer

mounted tangentially in the rim. The wheel electronics consists of remote start-up electronics, and pickoff and suspension electronics. The former provides the required signals for spin up, orientation and damping. The latter comprises the circuits which provide desired suspension level and control wheel suspended position. The accelerometer output feeds a resolver, integrator, sampler and stator control mechanism. The resolver rotor is mounted on a separate shaft which is slaved to follow twice the wheel angle. In this way reaction torques in the resolver do not disturb the wheel. An indication of the wheel angle is provided by a pencil light beam and photocell which generate a timing pulse twice during each wheel revolution. Mirrors are used to focus a light pulse on a photosensitive detector for every half revolution of the wheel. The pulse is produced as the light from the mirror passes through a slit located in a cover which surrounds the wheel assembly. A transmitter and power supply for telemetering data from the wheel are included.

A four pole inductosyn is used as the angular pickoff device to compare the electrical phase of the sensor output signal with the mechanical phase of the wheel rotation. The inductosyn electrostatically couples the rotor and stator and provides an output signal whose amplitude is proportional to the sine of the angle of rotation of the rotor relative to the stator.

A stator control mechanism uses the phase error signal to drive a motor which positions the stator. A digital shaft encoder is used to measure the angle of the stator with respect to a body reference axis.

Spacecraft equipment includes on-board attitude sensors, a sequencer, a logic and time reference package and a power conversion and filter unit. A horizon sensor and a star tracker are assumed as the on-board sensing devices used for vehicle orientation and in establishing the direction of the vertical for reference. A discussion of the technique of using a single star tracker as the master reference for this test is given in Appendix A.

Test Method - The wheel will be caged during boost into orbit but will be supported electrostatically during the test. Upon command, the wheel will be gently torqued until it reaches the desired test speed, but it will not be torqued during the test. The high inertia of the wheel and the frictionless suspension insure a reasonably constant speed. The testing sequence will be initiated by a ground command to align the vehicle pitch and roll axes in the horizontal plane by using the horizon sensor. The vehicle would also be coursely aligned in yaw by sensing the orbital angular rates in a "gyro compassing" mode. From the ground a star is selected which will be visible at some future point along the trajectory and will lie approximately in the orbit plane. The azimuth and elevation of the star with respect to the future vehicle attitude are computed and sent to the star tracker together with a timing command to tell the tracker when to begin search. When the vehicle arrives at the predicted position, the star tracker begins search at the precomputed azimuth and elevation. At lock-on, the vehicle is yawed through the small angle required to locate the star in the plane of the wheel and simultaneously null the azimuth gimbal angle. A zero angular rate control mode is then initiated, and the vehicle attitude during the subsequent test remains approximately the same as its instantaneous attitude at the beginning of this mode. The gimbal angles of the star tracker are telemetered to earth for use in computing the true direction of the vertical in vehicle coordinates. This true direction can then be compared with the direction indicated by the gravity-gradient instrument. After about five minutes of taking measurements, the test could be repeated or different wheel speeds could be commanded to obtain additional data.

Major Error Sources - Errors are introduced by random noise in the accelerometer signal which has frequency components in the pass band of the phase error detector. The phase error detector is effectively an extremely narrow band pass filter that drastically attenuates all signals except those at the desired signal frequency $2\omega'$. The bandwidth of this filter is proportional to $1/N$, where N is the ratio of the total integration time T to the period T_S of the wheel. For most orbital applications, the angular rate $\dot{\Phi}$ of the true vertical cannot change suddenly, so that a narrow bandwidth, about $2\omega'$, will pass the desired signal frequencies. All other noise is effectively filtered. Present accelerometer knowledge suggests that a noise power level of $\Phi_N = \frac{(10^{-10}g)^2}{|g_\theta|\ 2}$ can be achieved at some frequency $2\omega'$ when ω' is the angular rate of the wheel with respect to the vertical. From Reference (1), the corresponding error in the vertical is

$$e_\Phi = \sqrt{\frac{P_{out}}{P_{sig}}} = \left[\frac{\omega_{BW}\Phi_N}{|g_\theta|^2}\right]^{1/2}$$

The magnitude of the gravity-gradient signal is given by

$$|g_\theta| = \frac{3}{2} \cdot \frac{(GM)}{R^3} \cdot r$$

where r is the wheel radius, GM is the gravitational constant times earth mass, and R is the distance from earth's center. At an orbital altitude of 0.1 earth radius, with a wheel radius of r = 10 cm., $g_\theta = 1.77 \times 10^{-8}$ g's. With a time constant $1/\omega_{BW} = 100$ seconds, and assuming $N(2\omega') = \frac{(10^{-10}g)^2}{\text{rad/sec}}$, the error in the vertical is calculated to be

$$e_\Phi = .565 \text{ milliradians}$$

Acceleration components Δg_θ due to the presence of local masses will also have a component at frequency $2\omega'$. For a mass dm at distance ρ from the proof mass, and at angle Φ from the plane of rotation, this component has a magnitude

$$\Delta g_\theta = \frac{3kdm}{\rho^3} \cdot \frac{r}{2(1 + \sin^2\psi)}$$

If dm is a one pound mass at $\rho = 1$ foot with $\psi = 0$, then $\Delta g_\theta = 1.60 \times 10^{-11} g$. This is equivalent to an error in the vertical of 0.5 milliradians. Careful design in eliminating nearby masses or precalculation to bias out their effects will greatly reduce this source of error.

Master reference accuracy requirements will be satisfied by horizon sensors, having an associated 1.0° error in roll and pitch for ordinary atmospheric anomalies, and by a star tracker having twenty seconds of arc accuracy in the plane to be measured.

2.3.4 Experiment Physical Parameters - Table 2.3-1 describes the size, weight, and power requirements for experimental components.

2.3.5 Data Parameters - Experiment data parameters are listed in Table 2.3-2. The information presented includes signal format, parameter range, sample rate, accuracy and pertinent remarks. Important parameter characteristics are:

a. Accelerometer output noise amplitude and shape.

b. Timing pulse time of occurrence.

c. Resolver rotor phase and amplitude.

d. Phase error detector output amplitude.

e. Resolver stator angle magnitude.

2.3.6 Vehicle Orbit and Attitude Requirements - In order to insure an essentially drag free environment, a minimum orbital altitude of 300 nautical

TABLE 2.3-1

GRAVITY GRADIENT SENSOR EXPERIMENT PHYSICAL PARAMETERS

SYSTEM COMPONENT	SIZE		WEIGHT (LB.)	POWER	
	VOLUME ($FT.^3$)	L-W-H (IN.)		PEAK (WATTS)	NOMINAL (WATTS)
Experimental Device					
Gravity Gradient Sensor					
Wheel Assembly	0.50	12 x 6 x 12	15.0		
Support Electronics	0.08	6 x 6 x 4	1.5	3.0	3.0
Support Equipment					
Power Supply	0.13	5 x 6 x 8	10.0	40.0	40.0
Signal Conditioner	0.02	4 x 3 x 3	1.0	5.0	5.0
Programmer	0.013	2 x 3 x 4	1.0	5.0	5.0
Total	0.74		28.5	53.0	53.0

miles is preferred. An upper boundary on the orbital altitude of approximately 500 nautical miles will be set to insure sufficiently large gravity gradient magnitudes and therefore ease accelerometer requirements for this initial test.

A circular or low eccentricity orbit is preferred for initial test. The maximum orbital angular acceleration (Φ max) = 2 $e\omega_o$ = .001 rad/sec, Φ max = 2×10^{-8} rad/sec^2, equivalent to an error in the vertical of 0.01 milliradian, is well within the expected accuracy of the experiment.

No preferred orbital inclination will be specified since this is not critical to the experiment. There may be some advantage to a 30^o inclination orbit for telemetering purposes, however.

As previously stated, constant vehicle attitude angles are desired with the vehicle initially locally level and the velocity vector in the orbital plane. Vehicle angular rates of .05 deg./sec. will not produce excessive error.

TABLE 2.3-2
GRAVITY GRADIENT SENSOR EXPERIMENT DATA PARAMETERS

DATA POINT	SIGNAL FORMAT	PARAMETER RANGE	SAMPLE RATE (BANDWIDTH)	ACCURACY	RECORD	ESSEN-TIAL	DESIR-ABLE	CON-TINUOUS
Gravity Gradient Sensor								
Accelerometer	Analog	$\pm 10^{-6}$g	5.0 K CPS		X	X		X
Timing Pulses	Pulse	.01 to 2 pps	Sample at pulse rate ↓		X	X		X
Resolved Acceleration	Analog	$\pm 10^{-8}$g		1%	X	X		X
Integrator	Analog	Phase Error ±5°		0.1%			X	
Phase Error Detector	Analog	dc		1%	X	X		X
Stator Angle	Digital	± 5° (11 bits)		±20 sec	X	X		X
Temperature	Analog	0 to 120°		±5°, 4%			X	
Guidance & Control Sensors								
Star Tracker								
Elevation Gimbal	Digital	±70° 18 bits	20 samples/sec	±20 sec	X	X		X
Azimuth Gimbal	Digital	±30° 18 bits	20 samples/sec	±20 sec	X	X		X
Temperature	Analog	0 – 140°F					X	
Horizon Sensor								
Pitch	Analog	±20°	20 samples/sec	±1/2°	X	X		X
Roll	Analog	±20°		±1/2°	X	X		X
Temperature	Analog	0 – 140°F	20 samples/sec					
Rate Gyros*	400						X	
Yaw Rate	Analog	± 1°/sec.	5 cps	2.5%			X	
Pitch Rate	Analog	± 1°/sec.	5 cps	2.5%			X	
Roll Rate	Analog	± 1°/sec.	5 cps	2.5%			X	
Temperature*	Analog	0 – 140°F						
Time Reference	Digital	21 bits	Sample at timing Pulse rate	0.01 sec (±1 bit)	X	X		X

*Spacecraft Parameter

2.3.7 Experiment Support and Data Handling Requirements

Experiment Support - Control of the experiment environment requires that temperature and pressure be maintained within certain constraints. For this experiment temperature must be held between 0 and 140°F.

Mounting considerations are important for this test. The center of the wheel must be located accurately with respect to the vehicle center of mass (cm), and preferably located at that point. The horizon sensor, star tracker, wheel spin axis and the stator require alignment to vehicle axes within 0.1 degree. Movable masses in the vicinity of the accelerometer proof mass must be eliminated.

Power requirements for performing the entire mission are given in Table 2.3-1. About 135.0 watt-hours of electrical energy are required for the total mission.

Five ground commands are required to operate the experimental package and are included in the experiment test sequence bar graph (Figure 2.3-8) in their order and time of occurrence. Should the automatic commands listed in the bar graph fail to function, an identical ground command operational sequence will be employed. The automatic sequence is discussed in Section 2.3.8.

Test repetitions will be controlled by ground command. Three different wheel speeds (ω_s) of 0.1, 1.0 and 10.0 radians per second will be evaluated. Five test runs at each wheel speed will be employed to obtain sufficient data samples for a statistical analysis of the results.

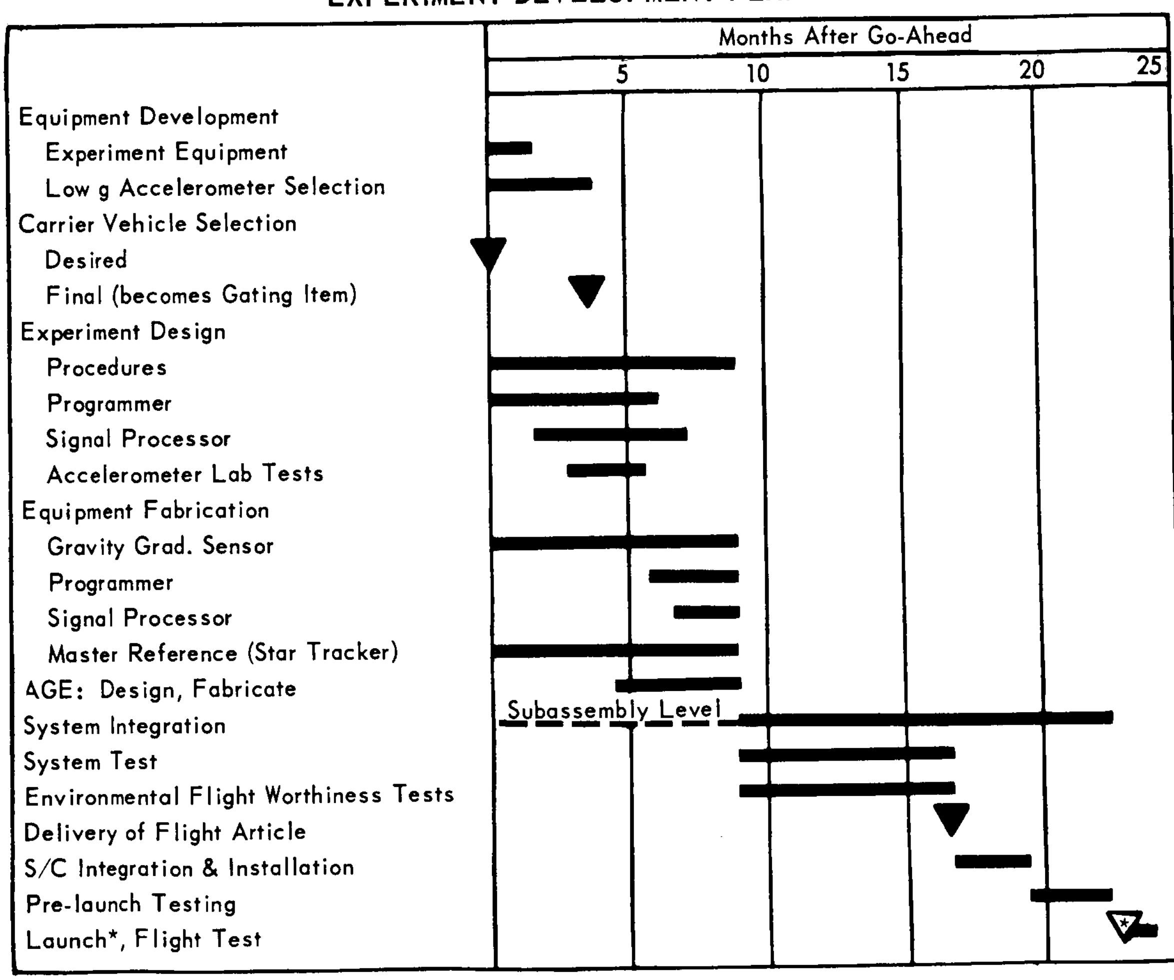
A discussion describing the use of a star tracker and horizon sensor as the experiment master reference is given in Appendix A.

Data Handling - General considerations for experiment data handling equipment may be found in Appendix B. Special attention should be given to the timing pulses derived from the wheel rotation and the analog output of the accelerometer.

Signal Processing of Accelerometer Output - The nature of the noise output of the accelerometer due to random accelerations is not adequately known. Definition of the upper frequency of this noise will allow the minimum sample rate to be determined. If the maximum noise output frequency is not predictable the data may be of such importance that a continuous channel is justified to handle the data.

Signal Processing of Timing Pulses - The timing pulses although bi-level in nature must be handled in such a manner as to preserve the timing of the leading

GRAVITY GRADIENT SENSOR
EXPERIMENT DEVELOPMENT PLAN



*Depends on Carrier Vehicle – may vary from 1 month to 8 months after installation.

FIGURE 2.3-7

edge of the pulse which is used as a time reference. Since the timing pulses will be asyncronous with respect to any spacecraft data multiplexing, continuous monitoring of the signal will be necessary. With proper isolation through a combining network the timing reference pulses could be superimposed on the accelerometer output channel. It is not necessary that the fidelity of the accelerometer output signal be preserved during the duration of the timing pulse.

A sequence, logic and time reference package consists of the necessary electronics, relays, memory circuits, time reference and logic networks to perform the following functions:

(1) Sequence data transmission to ground stations upon ground command.

(2) Conduct test upon ground command using either a ground command sequence or a programmed sequence.

(3) Monitor elapsed time during test.

(4) Sense failures of experimental package which would effect performance of the primary equipment and shut down the system if required.

2.3.8 Experiment Flight Test Plan - The anticipated schedule commencing with design of the device and culminating with its integration into the selected vehicle is shown in bar graph form in Figure 2.3-7. The entire effort is expected to require two years from go-ahead depending on the problem encountered in finding a suitable low-g accelerometer, problem areas requiring extension redesign and re-qualification, locating a satellite that can meet mission requirements, and unexpected launch delays.

The flight test program will consist of fifteen runs, including five runs at each wheel speed. Each run will consist of the events listed in the test sequence bar graph (Figure 2.3-8). These events are programmed to provide automatic in-flight command until a loss of tracking indication appears.

TEST SEQUENCE BAR GRAPH FOR GRAVITY GRADIENT SENSOR EXPERIMENT

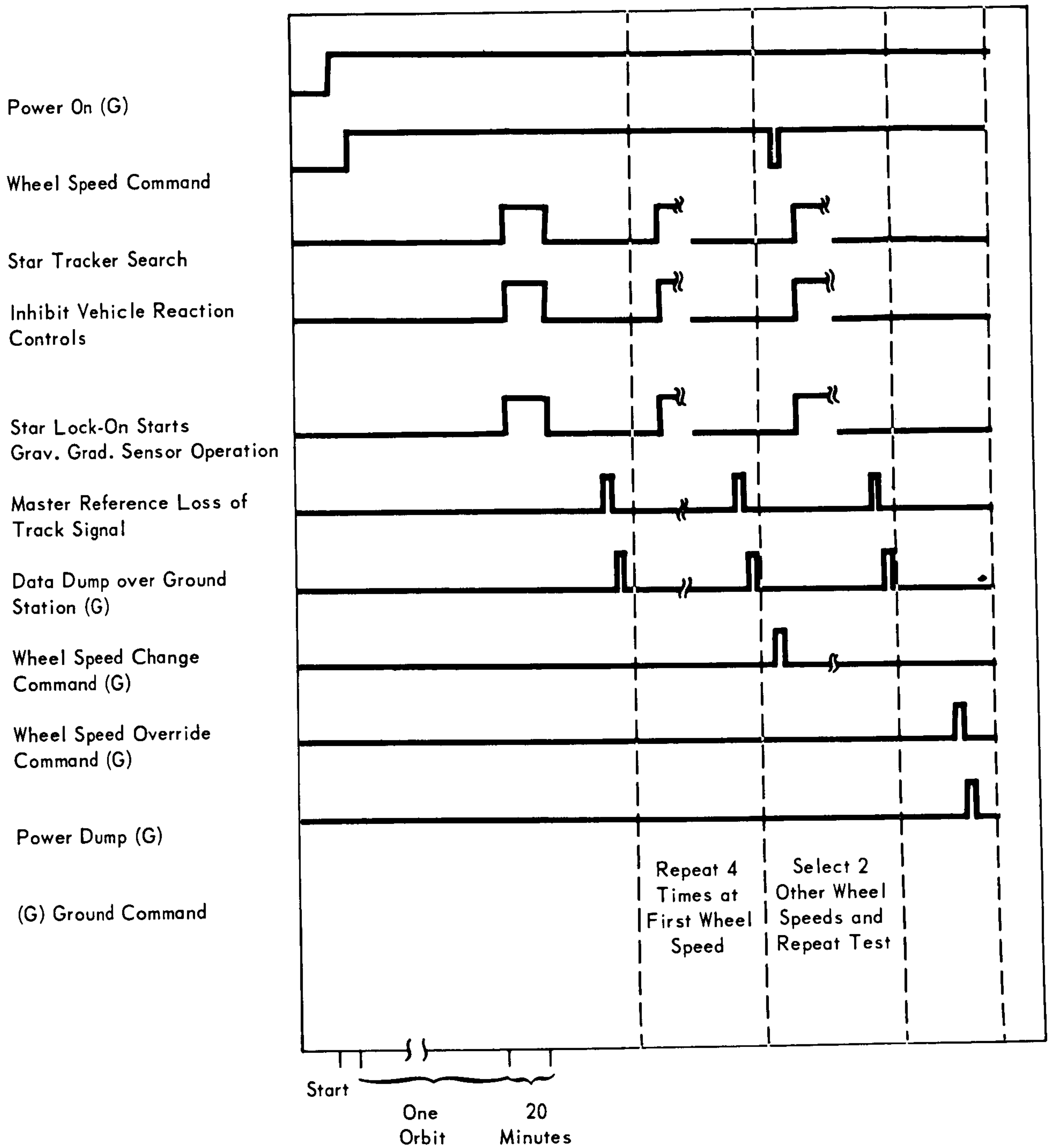


FIGURE 2.3–8

REFERENCES

1. Diesel, John W., "A New Approach to Gravitational Gradient Determination of the Vertical", AIAA Journal, Volume 2., No. 7, July 1964.

2.4 Earth Horizon Definition

2.4.1 Objective - The objective of this experiment is to obtain a precision measurement of the earth-space infrared (IR) and ultra-violet (UV) gradient characteristics in selected spectral bands. The experiment is a relatively long term test to obtain data on gradient position variation and slope variation as a function of season and day-night relationships. The scientific data derived from this experiment is to be used in establishing the ultimate accuracy of a precision horizon sensor. In addition, the data is to be used to determine the variables which should be included in horizon simulators to improve the validity of ground test evaluation of horizon sensors.

2.4.2 Background - In general, the accuracy obtained from existing horizon sensors is sufficient for antenna pointing and vehicle attitude control. For navigation and precise vehicle control, improved performance is required. Attitude sensing to better than 0.1° is desired for rendezvous and controlled re-entry. The horizon sensor uncertainties provide the major error sources in vehicle alignment for rendezvous and re-entry guidance of the Gemini spacecraft.

In order to design, build and test accurate horizon sensors a greater knowledge of the characteristics of the earth-space gradient is required. A recent report (Reference 1) indicates that performance to 0.05° is possible with an IR sensor if correct compensation is used. The compensation required is a function of the earth-space gradient variations and would require considerable analysis after precision data on the gradient is available.

There have been no major scientific experiments in orbiting vehicles to obtain precision measurements of the infrared (IR) and ultra-violet (UV) radiation characteristics of the earth. Horizon sensor design engineers need precision data obtained over a long time period and over the entire earth in order to establish the statistical nominal and maximum deviation of the gradient shape as a function of position and season. The data would be used to establish an optimum design for a precision horizon sensor. Nimbus and Tiros data covers a long time period but is not sufficiently accurate nor of the correct spectral pass bands to satisfy the requirements. The planned Project Scanner program is a short time mission using a sub-orbital spin-stabilized vehicle. Scanner data will be in two IR pass bands and will be high precision; however, the data will not include information on day-night variations, seasonal variations (Reference 2 and 3), tidal effects (Reference 4) and other factors which may cause gradient variations. Scanner measurements will be made to an altitude of 600 nautical miles during the 15 to 20 minutes of ascent and descent. The Scanner launch is to be from the east coast of the U.S. and will cover a very limited latitude and longitude range.

Horizon sensor design and ground test evaluation is limited due to the lack of precision scientific data on the radiation characteristics of the earth. Improving the design of precision horizon sensors is hampered by the lack of this information. Optimization of precision sensors involves the selection of the most stable spectral bands, sensing methods and techniques and the signal processing methods to be utilized to minimize the effect of gradient variations. Ground test equipment design is questionable since the amount and extent of anomaly or variation which should be used during a horizon sensor test is unknown. Each type of horizon sensor requires a simulator of different physical design because of the difference in field-of-view and sensing methods; however, all horizon sensors have common requirements for gradient simulation. For the common portions of horizon simulation (gradient slope, magnitude, anomaly and spectral characteristics) a design standard needs to be evolved. To establish the standard for simulators and to design precision horizon sensors, information is required in the following areas:

a. Precision measurement of the IR and UV gradient characteristics including slope, altitude, and stability as a function of latitude, seasonal variations and day-night variations.

b. Statistical evaluation of the size, shape and location of error-producing anomalies.

c. Narrow bandpass measurements of the gradient to establish the most stable operating band. Limited tests and considerable analysis indicate that the 14 to 16 micron band, the 20 to 35 micron water band and the 2000-to 4500-Angstrom UV bands look promising. Additional precision tests and analyses are required to determine the feasibility of using each of these bands.

Once the IR and UV radiation characteristics of the earth are precisely measured and determined, horizon sensor ultimate accuracy and design optimizing techniques may be established. In addition, variation parameters for horizon simulators may be determined and a better ground test of horizon sensor performance established.

Reference (1) provides a detailed analysis of the problem areas and discusses the parameters which must be taken into account in designing a precision IR horizon sensor. A summary of the major problems and the simulation difficulties for IR sensors are covered in the following paragraphs:

a. Temperature - The earth is an infrared radiating body with a black body equivalent temperature of from 160°K to 300°K, depending upon the spectral pass band of the horizon sensor detection/optics system, season, time of day, etc. For a pass band of 14-16 microns, temperature figures of 165°K to 265°K are normally used. Space background temperature is normally taken to be 4°K. Simulation using these temperatures is difficult and expensive and is not normally done on a day-to-day basis. Laboratory

CALCULATED HORIZON RADIANCE CURVES FOR A 14 TO 16 μ BANDPASS

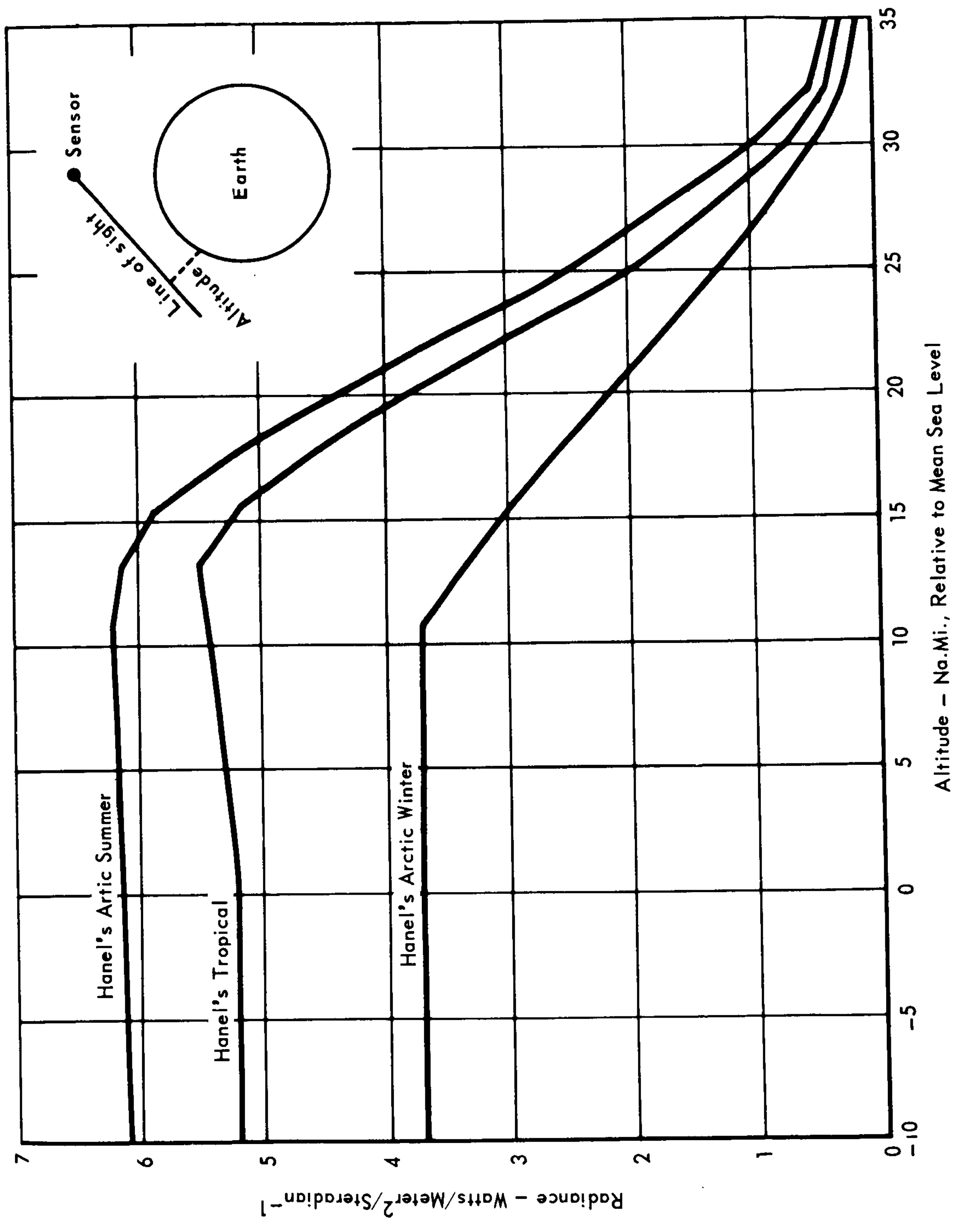


FIGURE 2.4-1

tests are usually performed using an ambient temperature for "space" and a hot surface for "earth". The differential temperature for simulation purposes is calculated on the basis of difference in radiant energy from the two sources. This technique is assumed to be satisfactory for most laboratory evaluation tests.

b. Gradient Slope - The simulation techniques used in the laboratories do not normally take into account the shape or slope of the earth-space IR gradient. There is very little known about the precise slope of the gradient. Papers (Reference 2 and 5) have been written which "define" the gradient for several spectral pass bands, the curves being based on meager non-precision data. The theoretical slope, amplitude, shape and breakpoint for the curves cover a wide range (Reference 2). Precision measurements are needed prior to undertaking the design task of slope simulation in horizon sensor test facilities. Figure 2.4-1 is Hanel's calculated curves of radiance as a function of altitude for the 14 to 16 micron band pass region.

c. Cold-Cloud Effects - Data from other programs indicate that if a horizon sensor used the 15 micron CO_2 band, the cold cloud effect would be smaller than variations due to climatic and seasonal differences in the earth's atmosphere. If, however, the sensor spectral pass band is wider than the 14-16 micron pass band, a realistic cold cloud simulation must be included in the evaluation tests. Detailed, earth coverage and a statistical analysis of the percentage of time that cold clouds exist in a nominal orbit would be required to design the simulator.

d. Seasonal Variations - Figure 2.4-1 clearly shows that the gradient position at a radiance level of 3 watts-meter^{-2}-steradian^{-1} varies approximately 8 n.m. between arctic winter and arctic summer, a variation of 48%.

The curves further indicate a maximum radiation variation of approximately 2.7 watts-meter^{-2}-steradian^{-1} between arctic winter and arctic summer, a variation in energy level of 75%. An ideal simulator for detailed laboratory evaluation would have to take the slope position variation and the energy level variation into account. Statistical knowledge is required to determine the exact amount of simulation variation required.

e. Day-Night Variations - It is generally assumed that the proper choice of optics and spectral pass band sufficiently reduces the effect of day-night variation. It may be found, however, that some "tidal" effects of the atmosphere may be present, and therefore the position of the gradient may vary as a function of time of day and season (Reference 4).

A general survey of test techniques used by IR horizon sensor manufacturers and users indicates that the simulation used for ground testing is only marginally satisfactory. Flight results indicate accuracies an order of magnitude less than expected (Reference 6), e.g., a 0.1° sensor provides a reference accuracy of about 1°. Designers indicate that ground simulation and system evaluation could be vastly improved if sufficient data were available on which to base precision simulator design.

Because of the difficulties experienced with infrared sensing techniques, the Instrumentation Laboratories of Massachusetts Institute of Technology has been investigating the possibilities of using the short wave UV reflected radiance from the earth. This would limit operation of the sensor to the daylight side of the orbit; however, with the projected improved performance of gyros, periodic measurements of attitude for platform update would be sufficient. The earth's atmosphere appears opaque at short wavelengths, meaning that the sun's energy is almost totally absorbed or reflected. This means that conditions near the surface of the earth do not add disturbing reflections. Radiation detectors operating at short wavelengths are more sensitive than IR detectors. The design of a local vertical UV sensing instrument requires that more knowledge be obtained on the exact characteristics of the short wavelength radiation.

MIT has proposed (Reference 7) that a complete study of the characteristics of the UV phenomena be performed from an orbiting vehicle. The primary factors to be measured are effect of sun angle, seasonal and latitudinal variations, spectral characteristics and determination of the altitude.

2.4.3 Functional Description - The major requirements for an accurate determination of the position of the earth-space gradient are:

a. Precision small field-of-view radiometers.

b. Precision measurement of the radiometer scan angle.

c. Precision measurement of the vehicle attitude.

d. Precision mounting of the experimental package components with respect to each other and with respect to the vehicle axes.

The experiment is designed to function around state-of-the-art hardware, using essentially off-the-shelf hardware where possible. A block diagram of the proposed experiment package is presented in Figure 2.4-2. The radiometers are used to obtain the gradient information. The star-tracker and gyro provide the precision vehicle attitude reference required by the test. Data storage while not over a tracking station is accomplished through use of a tape recorder.

Experiment Equipment - The radiometers provide a precision measure of the gradient position with respect to their mounting bases. The units provide outputs which are a function of position and the radiant flux level.

For measurement of the IR gradient characteristics, small field-of-view radiometers of the type built by Santa Barbara Research for Project Scanner are used.

FUNCTIONAL BLOCK DIAGRAM
EARTH HORIZON DEFINITION EXPERIMENT

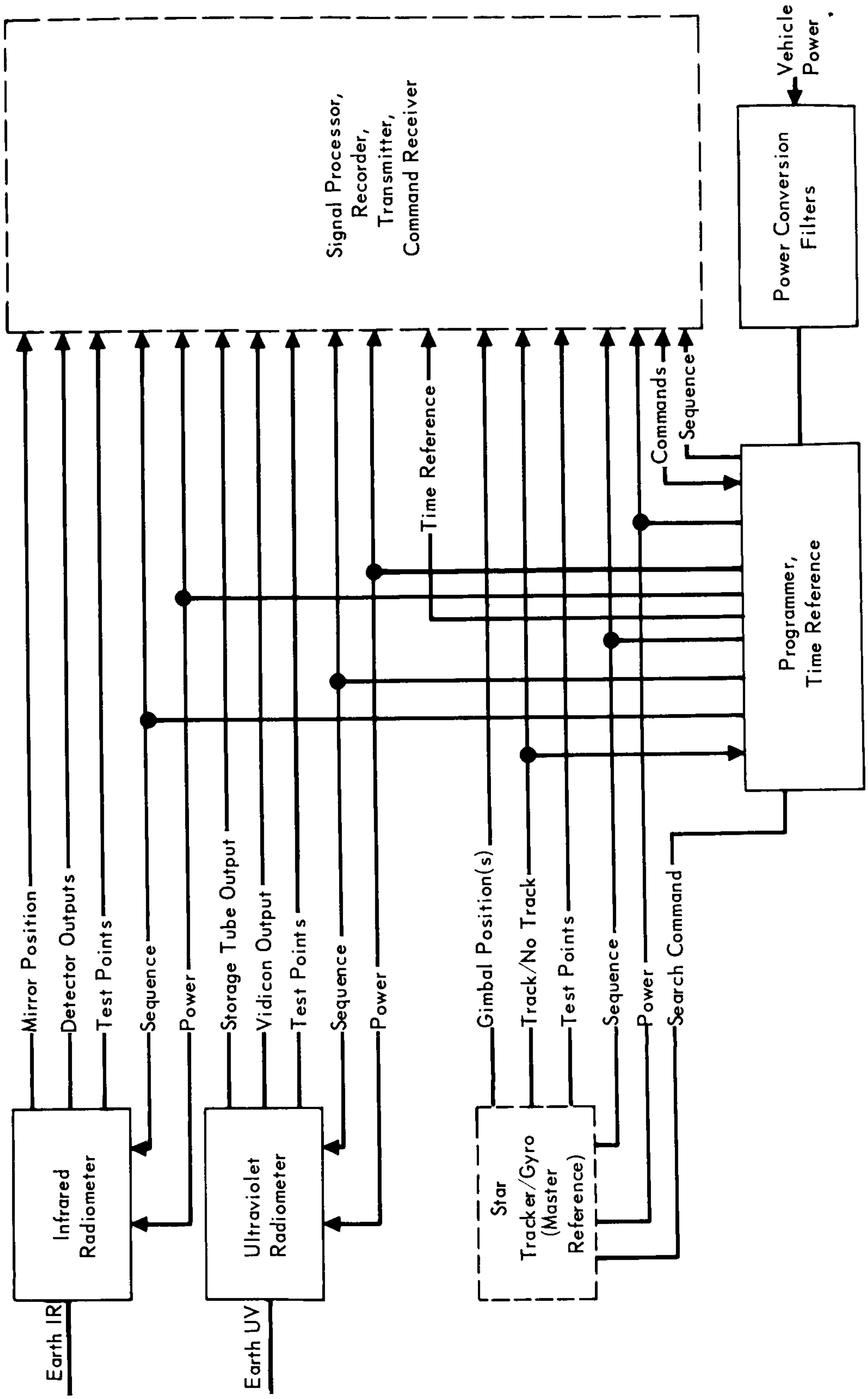


FIGURE 2.4–2

The radiometer has two sections and each section has a spectral bandpass filter, a scanning mirror and five 0.1° by 0.025° detector flakes. The flakes are positioned such that as the mirror rotates or scans across the gradient, the flakes detect the gradient change in sequence. This gives essentially five separate measurements of the gradient with each scan. The mirror scans through about 15° at a rate of 10° per second (linear saw-tooth motion). One section has a pass band of 14 to 16 microns (CO_2 band) and the other section has a 20 to 35 micron pass band (water vapor rotation band). The amplified detector outputs, scanning mirror position and selected test points are telemetered. Additional units with other pass bands may be added as room, power and needs dictate.

A UV radiometer is also included and operates in selected bands in the 2000 to 4500 Angstrom region. A representative unit has been designed by the Instrumentation Laboratory of MIT and consists of the optics, a vidicon, a storage tube, switching electronics and electronics associated with the tubes. A shutter is included as a part of the optics to prevent exposure of the image vidicon to direct solar radiation. Due to the data retrieval problem (bandwidth), exact design details have not been worked out on the lines per frame of the vidicon. If the number of lines per frame is 400, only 0.6 frames per minute are possible through a 2 KC bandwidth channel. Reducing the lines per frame or increasing the bandwidth of the channel will increase the data rate. Additional study is required prior to initiating the final design and fabrication of the system (Reference 7). Storage tube output, camera tube outputs and selected test points are telemetered. A recorder is used in conjunction with the storage tube for data storage when not over a tracking station.

<u>Attitude Determination</u> - From the purely scientific viewpoint, the desired accuracy of attitude determination would be 0.001° or better. For analysis of

horizon sensor design and for simulator design purposes, an accuracy of attitude determination of 0.01° to 0.05° would be satisfactory. Techniques for attitude determination of varying accuracies are described in Appendix A, "Master Attitude Reference System Considerations". The technique using one star tracker and a two axis gyro is recommended for this experiment.

Test Method - The experiment is conducted with the vehicle controlled to an earth-orbit plane orientation. After orbital insertion, the star tracker master reference is programmed to point forward along the vehicle orbital path. By selective logic in the star tracker computer, the tracker will search a small field (3° to 5°) and acquire a star. The tracker system will be programmed to continue searching and not to acquire if there is more than one detectable star present in the search pattern. Once the system acquires or locks-on, the gyro gimbals are slaved to the star tracker gimbals. When slaving is complete, the star tracker breaks lock and repeats acquisition and lock-on with a second star. The gyro maintains the inertial position of the first star. The star tracker tracks the second star until the telescope-gimbal reaches a predetermined gimbal stop position to the aft of the vehicle. When this occurs, the star tracker gimbals will be slewed to the forward position and the entire sequence repeated.

The specific stars being tracked can be determined with the knowledge of the star tracker readings, the vehicle's approximate attitude (as established by the satellite's prime reference system) and the vehicle's position in orbit as a function of time. Once the specific tracked stars are established, the precise vehicle attitudes can be determined by ground computers.

Star tracker gimbal positions, track/search signals, input power and several selected test points are telemetered.

When the track signal is received from the star tracker, the radiometer and star tracker outputs and the time reference signals are stored on tape and later telemetered to the ground.

Major Error Sources - Figure 2.4-3 is a plot of the inaccuracy in determination of the IR gradient position as a function of altitude and error in vehicle attitude determination. As an example, at an altitude of 200 nautical miles an attitude measurement error of 0.1° (6 arc minutes) represents about 2 nautical miles error in gradient position. A gradient position error of 2 miles represents an inaccuracy of about $\pm$ 10% in position determination. To obtain $\pm$ 1% data requires an order of magnitude improvement to 0.01° (36 arc seconds).

Star tracker and gyro performance is an order of magnitude better than horizon sensor reference systems. The OAO star tracker has an overall accuracy better than 20 arc seconds. A gyro with a maximum drift rate of 0.1°/hr. may introduce as much as 120 arc sec of error in a 20 minute time period. Horizon sensor performance is probably on the order of 1° in pitch and roll (Reference 6). This indicates that the horizon sensor will provide at best an estimate of vehicle attitude whereas the star tracker and gyro will provide the precision required to give 1% measurements.

The major error sources of this experiment excluding gyro drift are tabulated in Table 2.4-1. The major errors are star tracker and gyro angle readouts, time reference, vehicle rate (affects data correlation) and determination of vehicle position in orbit. The errors due to rate, data correlation and vehicle orbit position can be reduced by improving the time base accuracy from 0.1 sec to 0.01 sec. A time base accuracy of 0.1 seconds, assuming the timer is updated once every 8 hours, requires 19 bits. An accuracy of 0.01 second increases the number to only 22 bits which is a relatively small change.

ERROR IN DETERMINING GRADIENT POSITION AS A FUNCTION OF ERROR IN MEASURING VEHICLE ATTITUDE

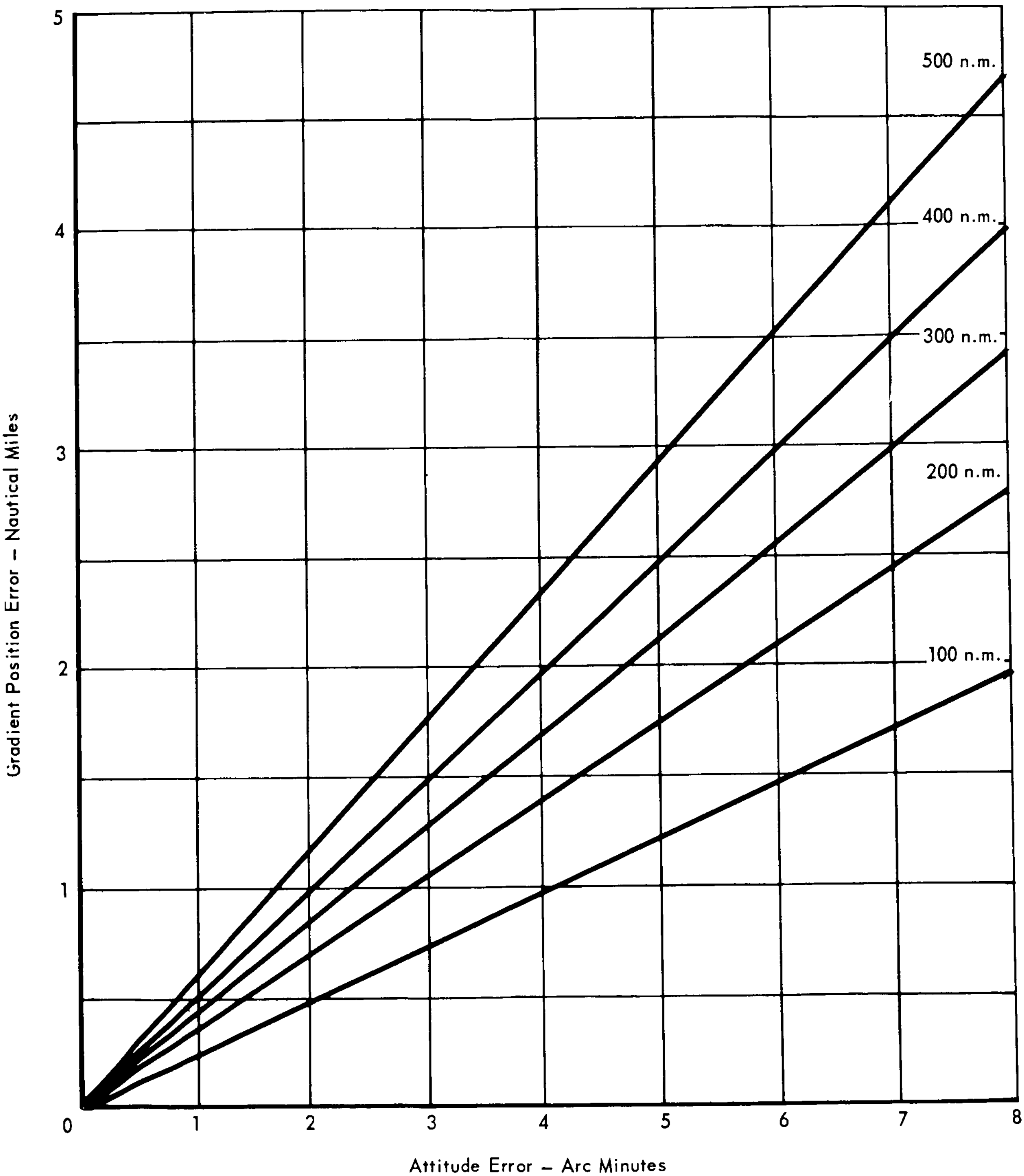


FIGURE 2.4-3

TABLE 2.4-1

EARTH HORIZON DEFINITION EXPERIMENT MAJOR ERROR SOURCES

ERROR SOURCE	TIME BASE ACCURACY	
	.01 SEC.	.1 SEC.
(Errors Given in Arc Seconds)		
Star Angle	0.1	0.1
Azimuth	0.1	0.1
Star Tracker Errors	20	20
Vehicle Rate Effect (.1° Sec. Accuracy)	3.6	36
Vehicle Orbit Position Determination	11	26
Radiometer Readouts	20	20
Data Correlation – 10%	5.5	10.22
RMS Error	34 Sec.	58 Sec.

Attainment of 1% performance, requiring measurements to 0.01 degrees, is possible using modified off-the-shelf hardware. Precision mounting of the radiometers, star tracker, and gyro with respect to one another is required. If the misalignment is a known quantity, data reduction compensation can be made. Of prime concern are mounting uncertainties and misalignments which occur during the launch phase and after orbital insertion (creep and temperature effects). Structural design and mounting should be such as to minimize these effects.

2.4.4 <u>Experiment Physical Parameters</u> - The physical parameters of the recommended equipment are provided in Table 2.4-2. The table contains a summary of the weight, size, and power requirements of the basic experiment package and the essential support equipment. The table also contains a summary of the requirements for the master reference star tracker system. This is included since there are no known possible carrier vehicles which are earth oriented and contain a star tracker

TABLE 2.4-2

EARTH HORIZON DEFINITION EXPERIMENT PHYSICAL PARAMETERS

ITEM	VOLUME		WEIGHT (POUNDS)	AVERAGE POWER (WATTS)	
	CU.FT.	L - W - H (INCHES)		PEAK	NOMINAL
Experimental Devices					
1. IR Radiometer	1.75	20 x 20 x 10	92	25.2	25.2
2. U.V. Radiometer	3	12 x 12 x 36	150	60	50
Support Equipment					
1. Program Sequencer Electronics	.03	5 x 3 x 3	5	10	10
2. Signal Conditioning Electronics	.06	5 x 5 x 4	7	-	-
3. Power Supply and Filters	.1	10 x 8 x 2	15	10	10
4. Recorder	.3	14 x 9 x 5	15	40	40
Subtotal	5.24		284	145.2	135.2
Master Reference* (less gyro)					
1. Star Tracker Head	.87	17 x 11 x 8	23.5	20	15.4
2. Star Tracker Electronics	.51	16 x 11 x 5	21.6		
Totals	6.62		329.1	165.2	150.6

*Required if not available on carrier vehicle

on board, although they are assumed to supply the master reference gyro. Estimates are given for the radiometer development effort based on a mid-to-late-1965 go-ahead.

2.4.5 Data Parameters - The information required from this experiment is a measure of the position, slope and amplitude of the earth-space gradient as a function of orbit position, time and spectral band-pass. On-board data is obtained from:

a. A precision measurement of vehicle attitude using star tracker/gyro techniques.

b. Precision measurements of radiometer mirror scan angle.

c. A measure of vehicle attitude as established by the earth oriented on-board attitude control system in order to process the star tracker data.

• Other vehicle operating parameters plus ground tracking for orbit determination are needed for data reduction and analysis.

Table 2.4-3 is a tabulation of the data parameters to be telemetered for this experiment. The priority or relative need for the signal is indicated in the remarks column by the notation desirable or essential. If the signal requires recording for data storage, it is indicated in the remarks column. Those signals

TABLE 2.4-3

EARTH HORIZON DEFINITION EXPERIMENT DATA PARAMETERS

DATA POINT	SIGNAL FORMAT	RANGE OF PARAMETER	SAMPLE RATE (FREQ. RESP.)	ACCURACY	RECORD	ESSENTIAL	DESIRABLE	CONTINUOUS	REAL TIME
I.R. Radiometer									
1. No. 1 Mirror Position	Pulses	15° in 3° increments	10/3 pulses/sec.	± .001 sec.	X	X		X	
2. No. 2 Mirror Position	Pulses	15° in 3° increments	10/3 pulses/sec.	± .001 sec.	X	X		X	
3. to 7. No. 1 Radiometer Outputs (5)	Analog	0–2 volts	80 cps	2%	X	X		X	
8. to 12. No. 2 Radiometer Outputs (5)	Analog	0–2 volts	80 cps	2%	X	X		X	
13. Radiometer Input Voltage	Analog	28 VDC	–	5%	X	X			
U.V. Radiometer									
14. Vidicon Output	Analog		2000 cps	2%	X	X		X	
15. Memory Tube Output	Analog		2000 cps	2%	X	X		X	
16. U.V. Radiometer Input Voltage	Analog	28 VDC	–	5%	X	X			
Star Tracker									
17. Gimbal No. 1	Digital	17 bits	20 time/sec.	± 1 bit	X	X		X	
18. Gimbal No. 2	Digital	17 bits	20 time/sec.	± 1 bit	X	X		X	
19. Input Power	Analog	28 VDC	–	5%	X	X			
Miscellaneous									
20. Time Reference	Digital	22 bits	20 time/sec.	± 1 bit	X	X		X	
21. I.R. Radiometer Temperature	Analog	0°F to 100°F	1 time/sec.	± 5%	X		X		
22. U.V. Radiometer Temperature	Analog	0°F to 100°F	1 time/sec.	± 5%	X		X		
23. Star Tracker AGC Voltage	Analog			± 5%					X
24. Star Tracker Gimbal No. 1 Torquer	Analog			± 5%					X
25. Star Tracker Gimbal No. 2 Torquer	Analog			± 5%					X
26. Star Tracker Track Indication	Digital	on/off	–	–					X
Required Spacecraft Signals									
* 1. Horizon Sensor Roll	Analog	0 – 5 V	5 cps	± 2%	X	X		X	
* 2. Horizon Sensor Pitch	Analog	0 – 5 V	5 cps	± 2%	X	X		X	
* 3. Horizon Sensor Loss-of-Track	Digital	On/Off			X	X		X	
* 4. Attitude Gyro Pitch	Analog	360°	5 cps	± 1%	X	X		X	
* 5. Attitude Gyro Roll	Analog	–80° to +80°	5 cps	± 1%	X	X		X	
* 6. Attitude Gyro Yaw	Analog	–10° to +10°	5 cps	± 2%	X	X		X	
* 7. Roll Rate Gyro Output	Analog	0 to ± 2°/sec.	5 cps	± 2%	X		X	X	
* 8. Pitch Rate Gyro Output	Analog	0 to ± 2°/sec.	5 cps	± 2%	X		X	X	
* 9. Yaw Rate Gyro Output	Analog	0 to ± 2°/sec.	5 cps	± 2%	X		X	X	
*10. Spacecraft Main Power Buss	Analog	20 to 30 VDC	60 cps	± 1%	X		X	X	

marked "real time only" are available upon ground command for system analysis. Additional test points for real time transmission may be added as needs dictate with an increase in the signal processing and data handling package being required.

2.4.6 Vehicle Orbit and Attitude Requirements - Table 2.4-4 contains a summary tabulation of the most desirable orbit and vehicle stabilization requirements. A discussion of the effect of variation of the parameters is contained in the following paragraphs.

Altitude - As indicated in Figure 2.4-3 the error in gradient position measurements is a function of altitude for a given attitude measurement error. As vehicle orbit altitude increases, the error increases. Ideally the vehicle altitude should be less than 200 nautical miles to minimize the error.

Eccentricity - The error is a function of altitude, therefore the eccentricity of the orbit should be as small as practical. An altitude variation of 300 nautical miles doubles the error in measurement of the gradient position.

TABLE 2.4-4

EARTH HORIZON DEFINITION EXPERIMENT ATTITUDE AND ORBIT REQUIREMENTS

PARAMETER	DESIRED CONDITION
a) Altitude	100 to 300 Nautical Miles
b) Eccentricity	Less than 0.014
c) Orbit Inclination	Polar (greater than 70°)
d) Stabilization	Earth Oriented in three axis Vehicle control deadband less than 2° Vehicle rates less than .1°/sec. Alignment to orbit plane – better than 5° in yaw.

Inclination - To obtain a maximum amount of data in the shortest calendar time possible, a near polar inclination is desired. This type orbit will provide in each orbit a winter arctic, equatorial and summer "arctic" gradient condition. Equatorial orbit inclinations would provide the least information on variables in a given period of time in orbit.

Vehicle Stabilization - The vehicle must be earth oriented and stabilized in three axes. If the vehicle utilizes horizon sensors for establishing local vertical and either a strap-down gyro system or a platform, the vehicle alignment can be expected to be better than 1° in pitch and roll and 5° in yaw. Operating the vehicle with a small deadband (less than 2°) and with low rates (less than 0.1°/sec) will provide a sufficiently stable "platform" for this experiment. Increasing the deadband or rates or decreasing the alignment accuracy will degrade the accuracy of the experiment. Any improvement in attitude control (increase in alignment accuracy, reduced deadband or rates) will improve the accuracy of the test. Ideally, the vehicle should be stabilized at null for the experiment.

2.4.7 Equipment Support and Data Handling Requirements - This experiment requires a considerable amount of support equipment. The data handling requirements are such that a very heavy load is placed on telemetry; in fact, a separate recorder and transmitter system will probably be required. General requirements for attitude determination and data handling are discussed in the Appendices. A discussion of the special requirements of this experiment, such as signal conditioning, data recording, programmers and mounting are contained in the following paragraphs.

Signal Conditioning - To provide a signal format which may be readily stored in a recorder and provide good correlation for later data reduction, signal conditioning will be required on some of the signals listed in Table 2.4-3.

The mirror position signal of the IR radiometer requires special conditioning to provide the desired "better than 0.01°" performance. The position pulses from the radiometer are fed into an automatic-reset decimal counter. In addition a clock input is fed in. After each 10 pulses (indicating one cycle of the mirror) the decimal counter will restart its count. The output is a 13 bit digital output which is a precision measurement of the time duration between radiometer mirror position pulses referenced to the time base. This method provides a 13 arc second capability in determining mirror position.

EARTH HORIZON DEFINITION EXPERIMENT
BIT DATA PROCESSING BLOCK DIAGRAM

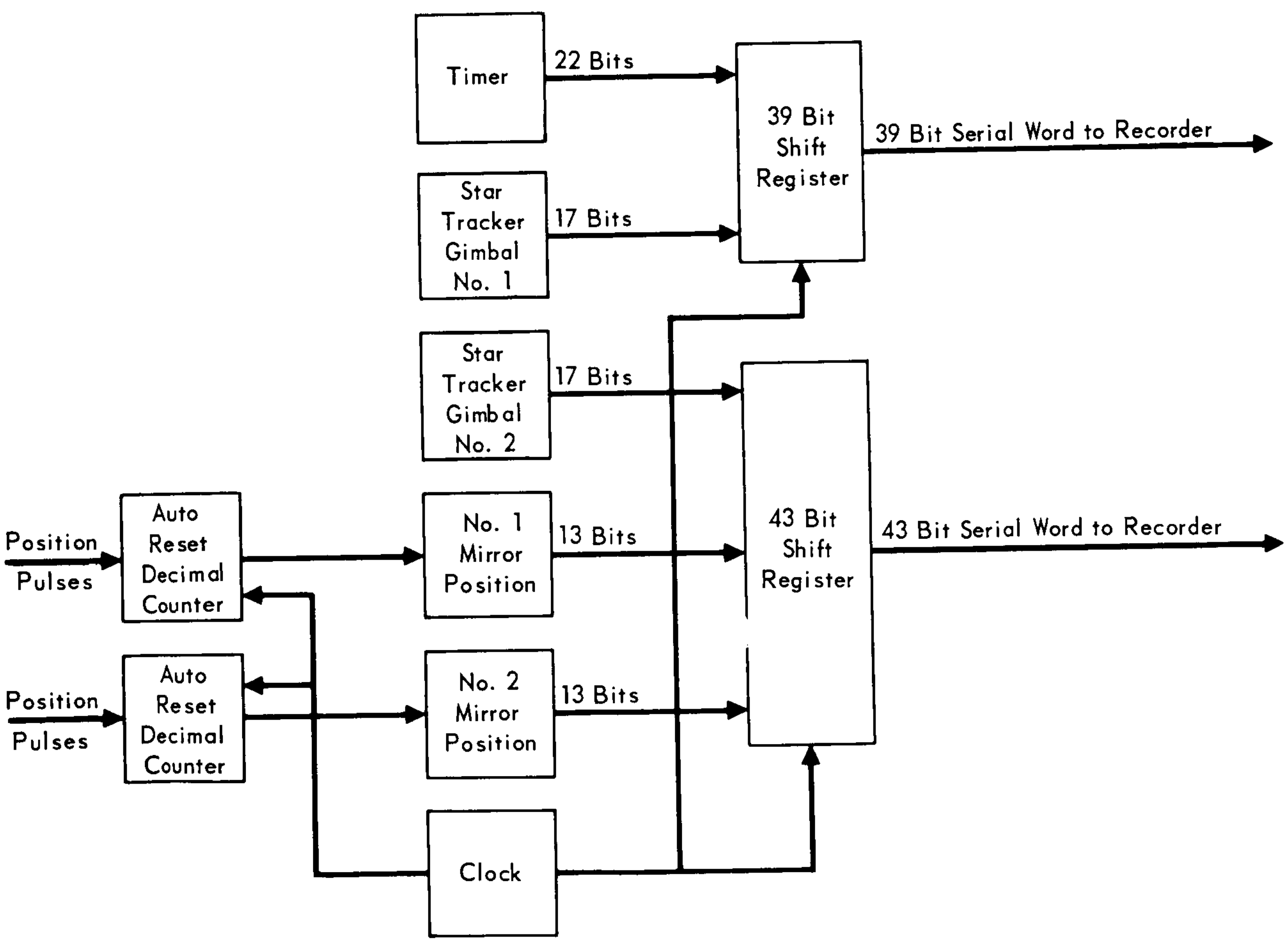


FIGURE 2.4-4

Star tracker gimbal positions and the time reference system signals are provided in digital format. Additional signal processing is required on these signals to provide recording. The conditioning consists of utilizing shift registers to yield serial words for recording. Figure 2.4-4 is a block diagram representation of the suggested method of serializing the bit words. The bits are serialized for recording on two channels of the tape recorder.

Master Reference - The recommended master reference system for this experiment is a single OAO star tracker and a two-axis gyro utilized as described in Appendix A.

Recorder - The recorder is required to obtain performance data from "around the world" in polar, equatorial and seasonal variation conditions. The selected data will be stored at a low tape speed and, upon ground command, will be dumped at a high tape speed to the ground tracking station. The maximum number of channels feasible should be taped in order that data reduction and performance evaluation is as unhampered by a lack of knowledge of parameter variation as possible.

Programmer - The programmer is specifically designed to serve the purposes required by this experiment. It consists of relays, memory circuits, time reference, logic and electronics to:

a. Turn system on an off either in a pre-programmed time sequence or upon ground command.

b. Sequence or command the Master Reference system to initiate search.

c. Turn recorder on during star tracking periods.

d. Update time reference system upon ground command.

e. Sequence data transmission to ground stations at pre-determined times or upon ground command.

f. Sense failures of experimental package which would affect performance of the primary vehicle and power-down the system as required.

EARTH HORIZON DEFINITION EXPERIMENT FIELD-OF-VIEW REQUIREMENTS

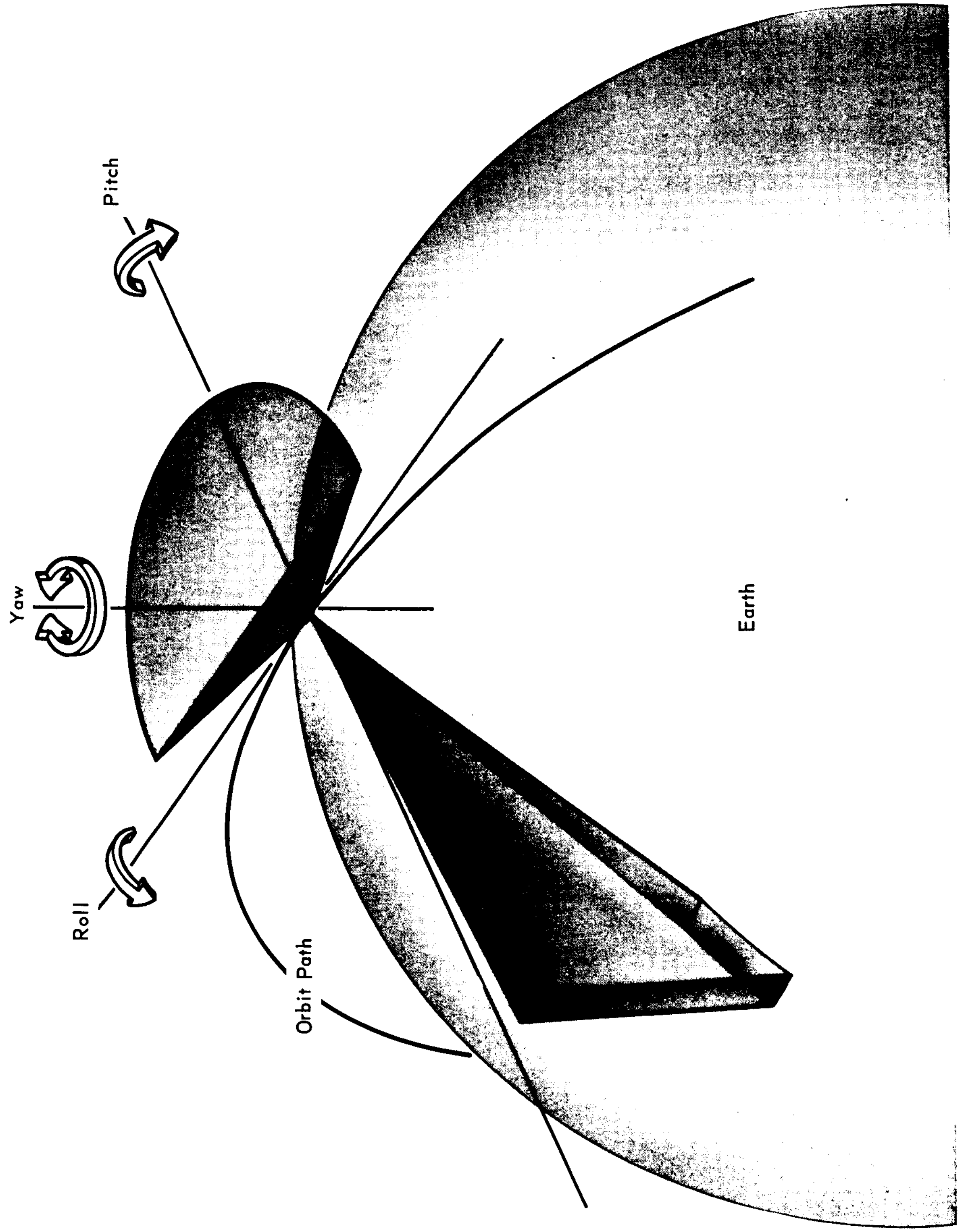


FIGURE 2.4–5

Environmental Control - The precision radiometers require stable operating temperatures for proper performance. To maintain the required 50°F to 90°F operating range, it may be necessary to add heaters to the radiometers. Good thermal bonding to the vehicle is essential. A thorough analysis of the heat load and thermal characteristics of the vehicle selected as the experiment carrier will be necessary to determine the needs.

Mounting - It is necessary to mount the radiometers in such a manner that an unobstructed field of view is available. The radiometers are mounted such that the plane of the field of view is perpendicular to the orbit plane, i.e., the pitch-yaw plane.

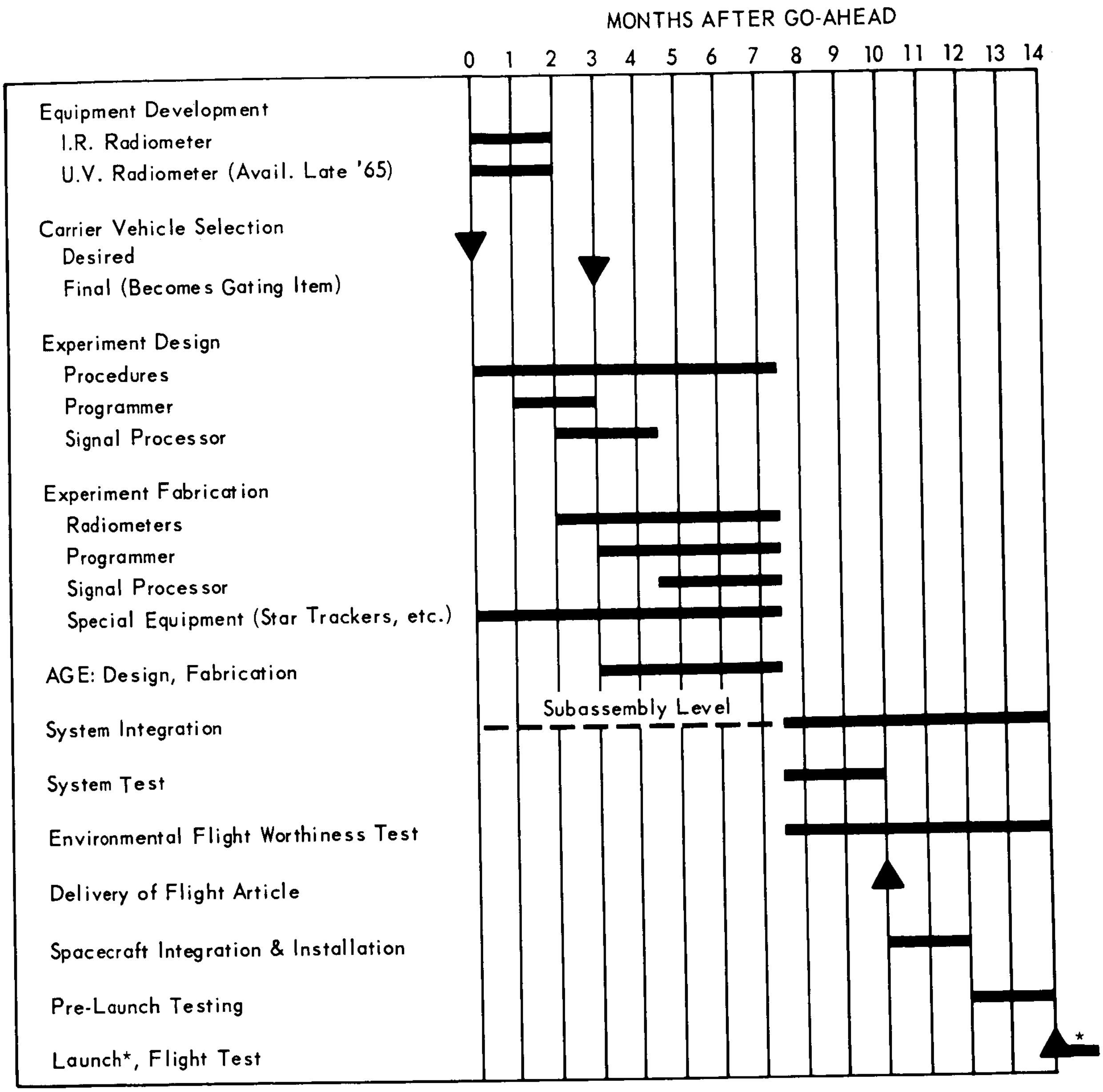
The master reference system also requires an unobstructed field of view. The OAO star tracker requires a field-of-view which is a cone of 150° (4.62 steradians) with the central axis of the cone parallel to the vehicle yaw axis.

Figure 2.4-5 is a drawing illustrating the field of view requirements of the IR radiometers and the star tracker. The field-of-view requirements of the UV radiometer is not illustrated since additional study is needed prior to establishing the required range.

Electrical Energy Requirements - The total power required for both the IR and UV experiment in a two-week mission with on-off cycles as described in paragraph 2.4.8 is 8,450 watt-hours excluding the energy requirements for the star tracker system. The total energy requirement including the star tracker, again based on the two-week mission with on-off cycles, is estimated to be 10,850 watt-hours.

Ground Command Requirements - The programmer is designed to function only partially on an on-board time base and requires periodic ground commands to update the sequence. Ground commands are necessary to:

EARTH HORIZON DEFINITION EXPERIMENT DEVELOPMENT PLAN



*Depends on Carrier Vehicle – May Vary from 1 Month to 8 Months After Installation.

FIGURE 2.4–6

a. Update the time reference system.

b. Command data "dump".

c. Revise operate time sequence.

d. Command real time transmission of troubleshooting test points when desired.

e. Command experiment system "on and off".

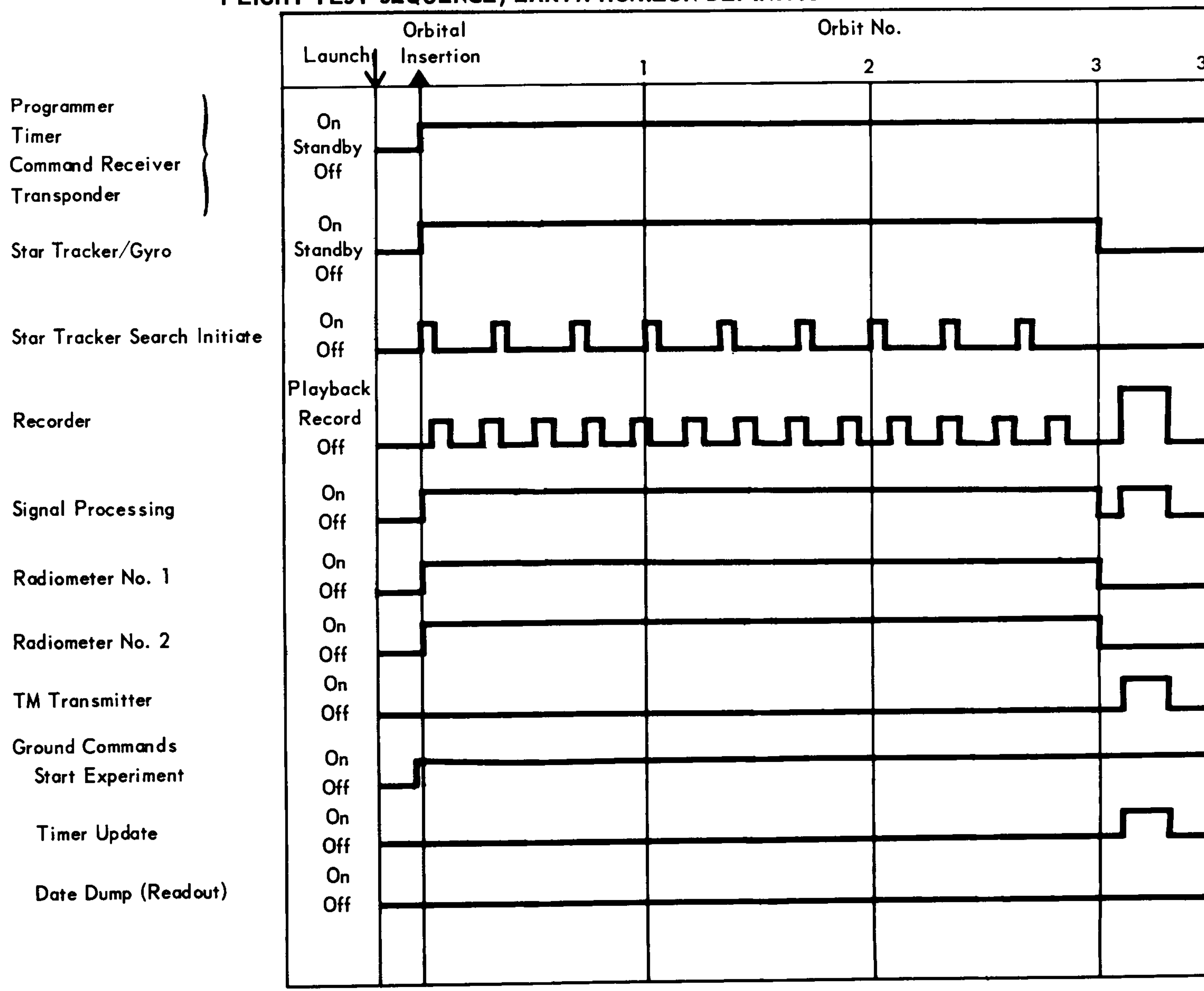
It is not expected that these five commands will cause any unusual loads on the ground station capabilities in a normal pass.

2.4.8 Experiment Flight Test Plan - Figure 2.4-6 is a flow diagram of the experiment development plan. The schedule is an estimate based on the required development, integration testing and flight time and is intended for planning purposes only. Areas which could cause delays or alter the schedule are programmer design, instrumentation design and U.V. Radiometer design and development. The IR Radiometer is developed (Project Scanner) and is not expected to present any schedule problems. The programmer design complexity is a function of the desired ground command override capability, desired or required redundancy and the requirements for supplying functions to the master reference system. The star tracker procurement time will be a variable depending upon the existing orders in work by the manufacturer. Systems integration and flight worthiness testing will require additional time if stringent radio interference (RFI) requirements exist.

The entire schedule is predicated on an early selection of a carrier vehicle; however, the fabrication of the master reference star tracker is the gating item of the development plan.

Flight Test Sequence - Figure 2.4-7 is a time plot of the sequences of the experiment for the first three orbits of the flight test. The program is based on acquiring a large amount of data over a fixed period of time. It is not necessary

FLIGHT TEST SEQUENCE, EARTH HORIZON DEFINITION EXPERIMENT



Note: See Figure 2.4–8 for Mission Cycles.

FIGURE 2.4–7

to operate the experiment continuously while in orbit. Because of the limited data storage capability of the recorder, it is more desirable to operate the experiment in on-off time sequences. Figure 2.4-8 is a time plot of the mission cycles for the suggested two week test period. The operation sequence is as follows:

a. After orbital insertion and vehicle stabilization the experiment package is turned on.

b. The star tracker is turned on and commanded to a predetermined angular position. The gimbal positions are based on the predicted position of a trackable navigation star as a function of the orbit path, vehicle orbital position, season and launch time.

c. After the star tracker acquires the first star, the gyro gimbals are slaved to the star tracker signals.

d. When slaving is complete, the star tracker breaks lock and gyro slaving is discontinued.

e. The star tracker is now commanded to another predetermined angular position as in step (b).

f. After the star tracker acquires the second star and starts tracking, the star tracker and gyro gimbal positions, radiometer outputs, time reference and other selected parameters are recorded.

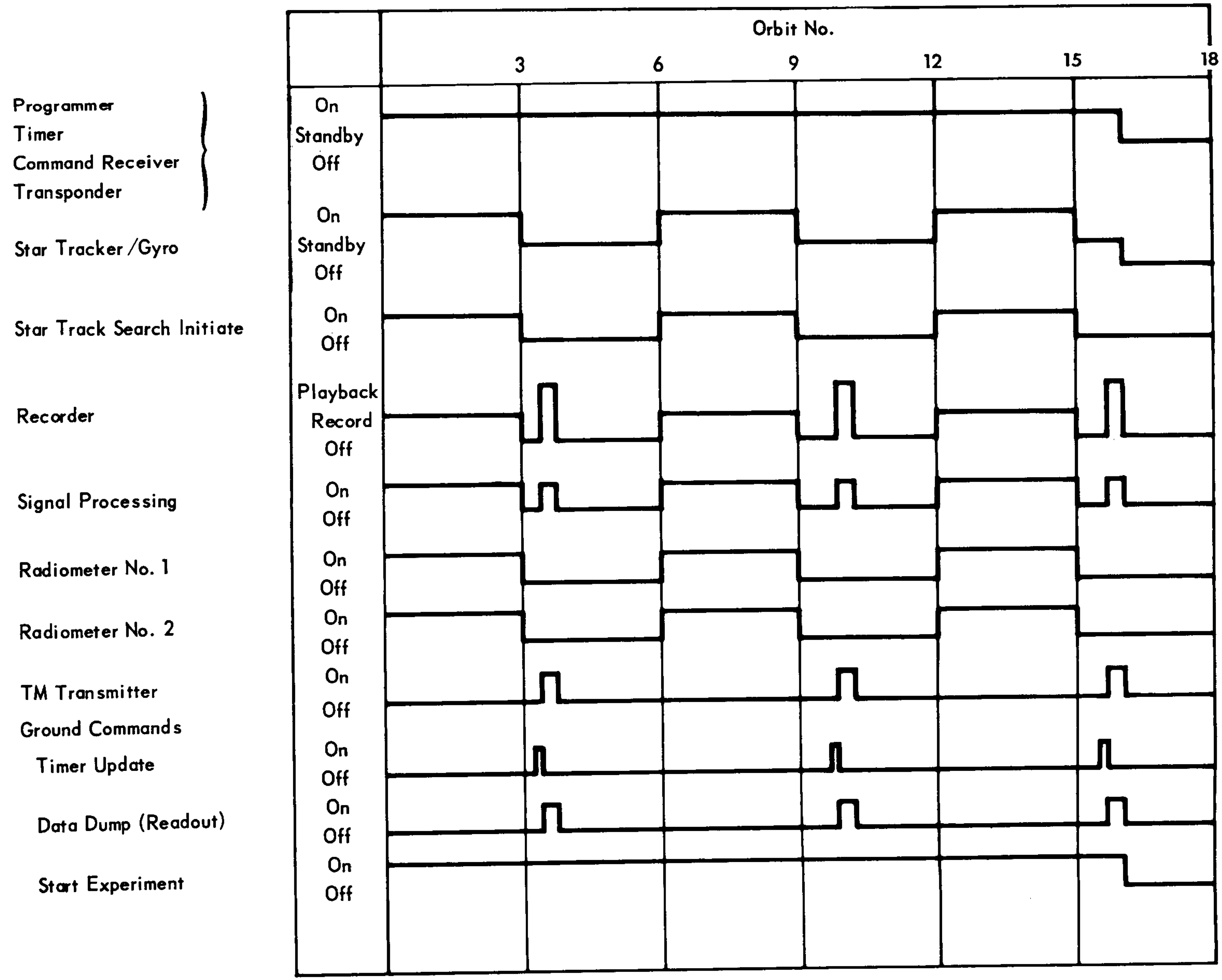
g. The recorder is operated for 5 minutes, then turned off for 15 minutes.

h. At the end of the 15 minute "off" period the recorder is turned on again if the star tracker is still tracking. If the star tracker is not tracking, steps (b) through (f) are repeated.

i. The cyclic recording of 5 minutes on, 15 minutes off is repeated for three orbits (270 minutes, 65 minutes recorded data).

j. The experiment shall be turned off for 3 orbits.

MISSION CYCLES, EARTH HORIZON DEFINITION EXPERIMENT



Note: Repeat above cycle during orbits 65 through 81, 131 through 147 and 197 through 213.

FIGURE 2.4–8

k. Steps (b) through (g) shall be repeated 3 times.

l. The experiment shall be turned off for 3 days and then the sequence of (b) through (h) repeated starting on the fifth day.

m. Repeat sequence (b) through (h) on the ninth and thirteenth day in orbit.

This sequence will provide at least 624 minutes of data, assuming the star tracker has lock-on at least 80% of the time. This will provide a nominal amount of data for a realistic evaluation of the nominal, extremes and statistical variation of the gradient as a function of time and orbital position. Information is obtained on seasonal (Northern hemisphere and Southern hemisphere are covered in each orbit), day-night and tide-effect variations over a wide range of latitudes and longitudes. Lengthening the test or increasing the recording time period will provide additional data for statistical analysis.

Post Experiment Data Reduction - Figure 2.4-9 is a line flow diagram of the data reduction process. The sequences are described in the following paragraphs:

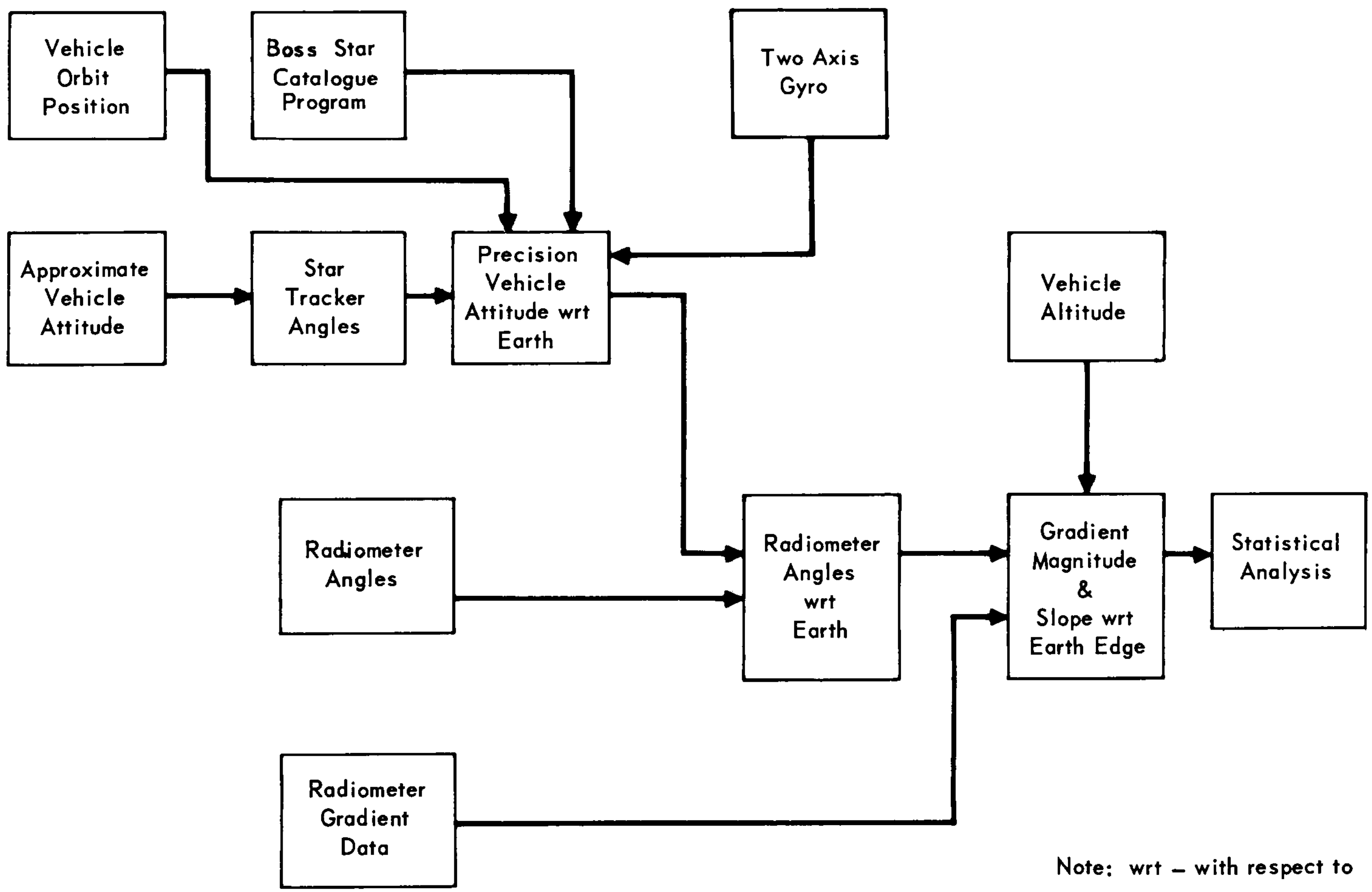
a. Determine, through use of the on-board horizon sensor and gyro reference system, the approximate attitude of the carrier vehicle with respect to earth local vertical.

b. Time correlate the vehicle orbit position obtained through ground tracking.

c. Through use of vehicle orbit position, approximate attitude, Boss General Star Catalogue and the star tracker and gyro angles during track, determine which specific stars are being tracked.

d. Once the specific stars being tracked have been established, precision vehicle attitude is computed. Using a single star tracker and a gyro will provide a measure of the vehicle local horizontal plane with respect to the earth reference axes.

EARTH HORIZON DEFINITION EXPERIMENT DATA FLOW DIAGRAM



Note: wrt – with respect to

FIGURE 2.4–9

e. From the vehicle altitude, compute the correction factors to be applied to radiometer outputs.

f. Apply the radiometer correction factors and compute the gradient parameters as a function of position with respect to mean sea level. The computation requires a knowledge of vehicle altitude as well as attitude.

g. Process the data to obtain the average and deviation of the gradient magnitude. The data will be a function of altitude, latitude, longitude, day-night and seasonal variations. The data should be so analyzed as to provide a statistical analysis of the effect of the variables on the gradient.

REFERENCES

1. "Ultra-High-Accuracy Horizon Sensor Study", Advanced Technology Laboratories, A Division of American-Standard, ATL-D-1225, July-September 1963.

2. R. A. Hanel, W. R. Bandeen and B. J. Conrath, "The Infrared Horizon of the Planet Earth" Journal of the Atmospheric Sciences 20 No. 2, March 1963, pp 73-86.

3. D. Q. Wark, J. Alishouse, G. Yamamote, "Variations of the Infrared Spectral Radiance Near the Limb of the Earth", Applied Optics, 3, No. 2, February 1964, pp 221-227.

4. S. T. Butler, "Atmospheric Tides", Scientific American, 207 No. 6, December, 1962, pp 48-55.

5. J. W. Burn, "The Application of Spectral and Spatial Characteristics of the Earth's Infrared Horizon to Horizon Scanners", IEEE Transactions on Aerospace-Support Conference Proceedings, pp 1115-1126.

6. Barbara K. Lunde, "Horizon Sensing for Attitude Determination", paper presented at the Goddard Memorial Symposium of the American Astronautical Society, Washington D. C., March 16 & 17, 1962.

7. Letter Enclosure, "Plan for Experimental Orbital Navigation Investigations from a Space Vehicle", Instrumentation Laboratory. Massachusetts Institute of Technology, January 5, 1965, 19 pages.

2.5 Horizon Sensor Accuracy

2.5.1 Objective - The objective of this experiment is to measure the in-flight accuracy of horizon sensor system determination of vehicle attitude with respect to earth local vertical. The data will be used to initiate or evaluate design improvements for horizon sensors and to determine the effectiveness of horizon simulation used in ground tests.

2.5.2 Background - The accuracy of a horizon sensor in establishing a local vertical for an orbiting spacecraft is uncertain. The precision performance obtained in ground tests has not been obtainable in flight. The difference in performance characteristics has been intuitively, and to some extent experimentally, evaluated. Variations in the earth-space infrared gradient characteristics, cold cloud formations and other variables have caused sensor performance to be considerably poorer than expectations. Lack of detailed knowledge of the earth as an infrared (IR) target has made it virtually impossible to build a precision simulator which realistically takes into account all of the various error producing anomalies. Each sensor design requires a different type of simulator due to the differences in field-of-view and sensing techniques. Therefore no one particular type of simulator design has been evolved to test all types of horizon sensors.

Simulator design parameters which must be taken into account are temperature, gradient slope, cold cloud effects, seasonal variations and day-night variations. Detailed discussions of the variables are contained in Paragraph 2.4.2, "Earth Horizon Definition Experiment". If sufficient knowledge were available on the earth's IR characteristics, a test facility could be designed for horizon sensors. The special design facility would provide a precision measure of the horizon sensor characteristics with most of the variables taken into account. It is not necessarily desirable to build a "universal" test facility inasmuch as the cost for a facility of this type would be quite high, and in addition it would have to be quite large. Horizon sensor optics are normally focused to infinity, and hence for a realistic simulation with only flight optics involved, the IR "target" would have to be no less than 16 feet from the sensor. To test all types of horizon sensors (conical scan, edge tracking, radiometric balance, etc.) at a simulated altitude of 80 nautical miles would require an earth target simulator 286 feet in diameter. The simulator and chamber wall would be cooled with liquid nitrogen and liquid helium. In addition to the design and fabrication costs, there would be the daily operation costs which could run several thousand dollars.

A low precision ($\pm 1^{\circ}$) cold cloud simulation facility, with no attempt made to simulate the gradient slope, was used on Project Mercury horizon sensors. The horizon sensor was placed in a 3 foot diameter high vacuum chamber which had surfaces cooled with liquid CO_2 (earth) and liquid nitrogen (space). A cooled variable size wedge provided the "cold-cloud" effect. Variations in cloud size required breaking the vacuum and manually rotating the wedge. The earth and cloud simulator for the chamber costs approximately $5,000 and daily operating costs for the facility were about $1,500. Although the test provided further understanding of the "cold-cloud" problem, the initial and operating costs were high,and the data and simulation was not sufficiently accurate to provide the information required for sensor redesign.

To determine the stability of the horizon sensor tracking loop over a wide range of simulated earth-space temperatures at various sensor head temperatures a test was performed on the Gemini horizon sensor. The test was conducted in a 16 feet long, 2 feet diameter stainless steel vacuum chamber owned by the Air Force. The horizon sensor and an earth-space gradient simulator were placed at opposite ends of the chamber. The earth temperature was varied over a range of 140 K° to

$250^{\circ}K$ in order that data on signal-to-noise ratio, gain and phase margin could be obtained. There was no attempt made to provide a simulation of gradient shape or slope, or to simulate cold clouds. The test cost approximately $15,000 and no information was obtained on the sensor accuracy characteristics. The only data obtained was tracking loop stability for a static (azimuth yoke stationary) condition. Again, as in the case of the Mercury tests, the data was extremely valuable, but results were very limited.

Providing realistic simulation could require that cold clouds, gradient slope, vehicle motion, altitude, attitude, solar simulation, temperature, emissivity, absorbtivity of the earth and several other factors be simultaneously varied in a realistic manner. Because of the lack of precision data on the earth space gradient and the extreme costs for a detailed simulation of all the calculated or theoretical variables, there is a real advantage in a detailed flight evaluation of a well designed, "optimized", precision horizon sensor. The flight test would yield information on the in-flight accuracy of a horizon sensor. To date, all accuracy evaluation has been based on an intuitive analysis of the horizon sensor using the horizon sensor itself as the "reference". Evaluation of the data from such a flight test might indicate that many of the IR gradient variations and other variables are insignificant to the performance of the sensor. A properly instrumented experimental package would also provide precision information on the effect of the orbital environment, IR gradient variation, vehicle dynamics and orbital parameter effects simultaneously.

2.5.3 Functional Description - The major requirements for an accurate measurement of the performance characteristics of a horizon sensor are:

a. Precise determination of vehicle attitude with respect to local vertical.

HORIZON SENSOR ACCURACY EXPERIMENT BLOCK DIAGRAM

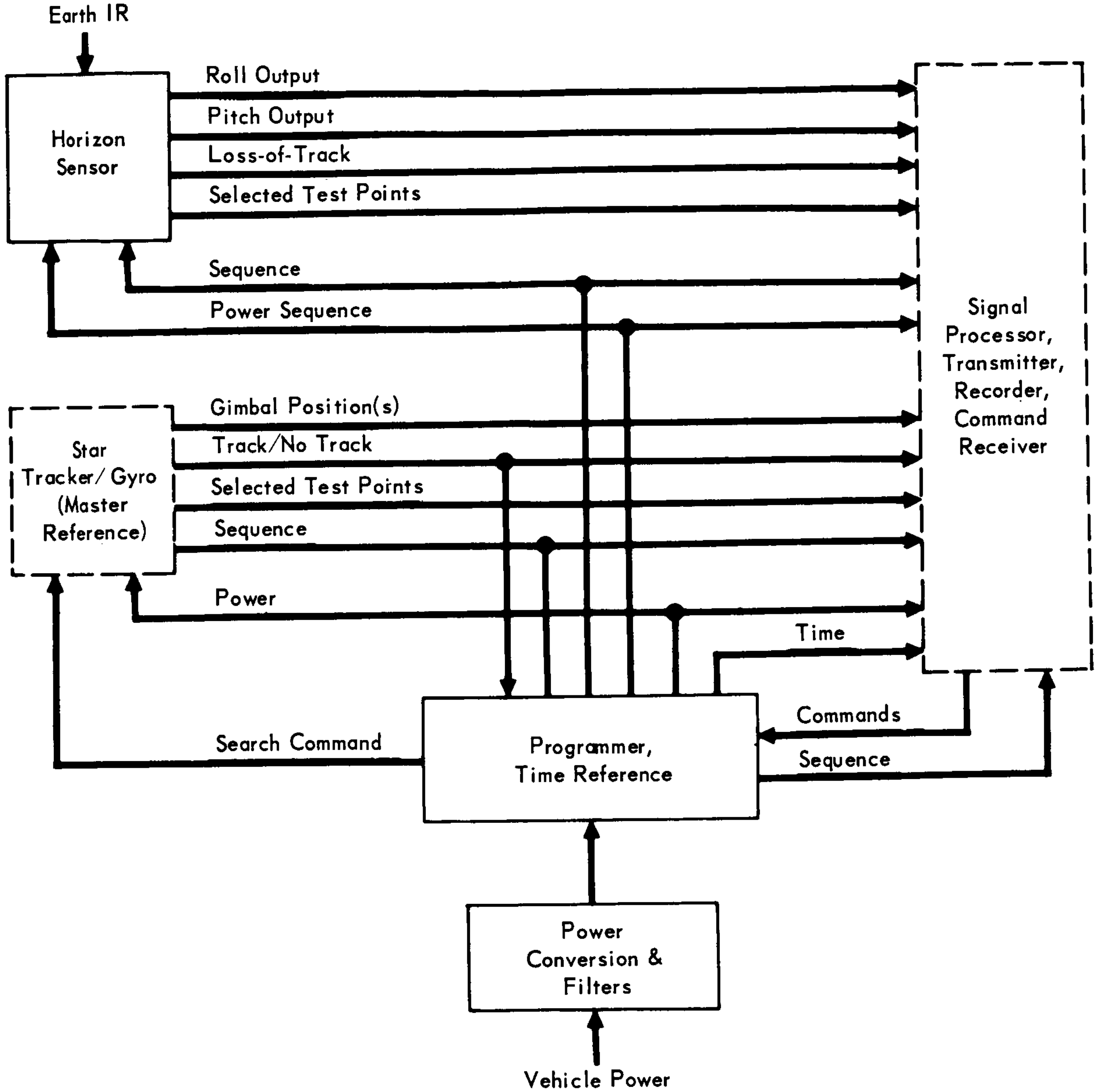


FIGURE 2.5-1

b. Precision mounting of the horizon sensor with respect to the master attitude reference and the spacecraft axes.

c. Precision data retrieval.

The experiment is designed to function around state-of-the-art hardware utilizing off-the-shelf hardware where possible. Information on the effect of seasonal variations, day-night variations, and systems operational parameter variations over a wide range of longitudes and latitudes is obtained. Close control of system operational parameters is maintained (or thoroughly monitored) in order that the effect of variables may be carefully evaluated.

A functional block diagram of the experimental package for measurement of a horizon sensor accuracy characteristics is shown in Figure 2.5-1. After orbital insertion and vehicle stabilization, the star tracker will acquire and track a star, providing star angle data in two axes. Horizon sensor outputs, master reference signals, time reference pulses and other selected parameters are recorded for data retrieval over tracking station sites. The recorder is programed to record either continuously or in sequenced on-off time cycles to provide a large statistical sample of horizon sensor performance.

Experiment Equipment - The techniques described in this experiment will provide an evaluation of any precision horizon sensor; however, to illustrate the design parameters requiring consideration, the Gemini horizon sensor is used as a model. Usage of other sensors will introduce some changes in the field-of-view requirements, the physical parameters and the signals to be telemetered. The sensor, built by Advanced Technology Labs, provides an output which is essentially an integration of several edge measurement samples over a wide area and hence anomaly caused errors are reduced as compared to some techniques. Modification to the Gemini unit should consist of changing the optics from an 8-18 micron spectral pass band to a 14- 16 micron spectral pass band. The modification is recommended in order that the more stable CO_2 band is used for evaluation purposes. Sensor pitch and roll outputs, loss-of-track indication, input power, temperature, position amplifier output, azimuth amplitude and other selected test points will be telemetered.

Attitude Determination - To evaluate the accuracy of a horizon sensor, a reference system with an "order-of-magnitude" greater precision than expected from the test unit is desired. Techniques for attitude determination with varying degrees of accuracies are described in Appendix A, "Master Attitude Reference System Considerations". The technique using a single OAO star tracker and a two axis gyro is recommended for this experiment.

Test Method - After orbital insertion and initial vehicle stabilization in an earth orbit plane, the star tracker is programmed to point forward along the orbital path. By selective logic in the star tracker programmer, the system will search a small field (3^o to 5^o) and acquire the unique or single detectable star in that field. The system is programmed to continue searching until the condition of only one detectable star in the search pattern is present at which time the system will be commanded to acquire. Once the system acquires or locks on the angle data is stored in a two axis gyro memory. The star tracker is then commanded to search, acquire and track a new star. When the telescope gimbal reaches a predetermined angle from aft of the vehicle, prior to the gimbal stop, the memory gyro is slewed to the star tracker position. The star tracker gimbals will then be slewed to the forward position and the search/acquire/track sequence repeated.

Using the knowledge of the star tracker angle to a star and the gyro stored star angle, the vehicle's approximate attitude (as established by the satellites prime reference system) and the vehicle's position in orbit as a function of time, the specific stars being tracked may be determined. Once the specific tracked stars are established, the precise vehicle attitudes can be determined.

Although the star tracker and gyro outputs are not a direct measure of the vehicle attitude with respect to earth, the earth reference information can be readily obtained. Using time as the baseline, the star angle information, the vehicle position in orbit and the star position with respect to earth, a precision measure of the spacecraft attitude with respect to the earth local vertical may be determined. The data on spacecraft attitude when compared with the horizon sensor outputs, again as a function of time, will provide the precision evaluation of a horizon sensor's capability to establish a measure of the earth local vertical for a spacecraft. It should be noted that the horizon sensor outputs, although normally referred to as pitch and roll outputs, are actually a measure of the horizontal plane position and the outputs are interdependent. This can best be illustrated by taking an example. If the spacecraft were pitched 5° with no roll or yaw reference misalignment, the horizon sensor would indicate a 5° pitch output. If the vehicle is then rotated about its own yaw axis, the pitch output would start decreasing and the roll output would start increasing from null. At a yaw angle of 90°, the roll output would be 5° and the pitch output would be 0°. In other words, the horizon sensor outputs are a measure of the attitude of the plane of the vehicle which is perpendicular to the yaw axis. Horizon sensor outputs provide a vector measure of the planar attitude with respect to vertical. The star tracker and gyro provides, after proper position correlation, a measure of the spacecraft attitude with respect to local vertical. A comparison of the spacecraft attitude as measured by the horizon sensor and the attitude as determined by the star tracker/gyro knowledge will provide a precision measure of the horizon sensor capability of measuring the earth local vertical. The technique described herein does not necessarily provide individual axis information; however, if the horizon sensor cross coupling characteristics are incorporated into the computer program, the individual axis data may be obtained.

Major Error Sources - The major error sources for this experiment are star tracker and gyro angle readout, gyro drift rate, time reference, vehicle rates (affects data correlation) and determination of the vehicle's position in orbit. As can be seen from Table 2.5-1, a tabulation of the major error sources of this experiment, improving the time base accuracy from 0.1 sec to 0.01 sec reduces the errors primarily associated with data correlation. If the time reference system is updated once each eight hours with an accuracy requirement of 0.01 second, a 22 bit clock word is required. Since an accuracy of 0.1 second under the same conditions requires a 19 bit clock word, the 0.01 second requirement adds no significant complexity. Table 2.5-1 does not include the gyro error sources.

Attainment of 0.05° accuracy for the reference system is possible using a single star tracker and a two axis gyro as the master reference. Of prime importance is the alignment of the horizon sensor to the master reference. Compensation for known measured misalignment can be performed in the data reduction process. Shifts or creep which occur during the launch phase or while in orbit cannot be determined. Care should be observed in the vehicle design and the

TABLE 2.5-1

HORIZON SENSOR ACCURACY EXPERIMENT MAJOR ERROR SOURCES

ERROR SOURCE	TIME BASE ACCURACY	
	0.01 SEC.	0.1 SEC.
(Error Given in Arc Seconds)		
Star Angle	0.1	0.1
Azimuth	0.1	0.1
Star Tracker Errors	20	20
Vehicle Rate Effect (.1°/Sec. Accuracy)	3.6	36
Vehicle Orbit Position Determination	17	26
Data Correlation – 10%	3.5	8.2
RMS Error	23.4	49.4

installation of the experiment to assure a minimum change in misalignment between the sensors.

2.5.4 Experiment Physical Parameters - The physical and operation parameters of the recommended equipment are provided in Table 2.5-2. The table contains a summary of the weight, size and power requirements of the basic experiment package. The table also contains a summary of the requirements for the star tracker portion of the master reference system. This information is included since there are no known possible carrier vehicles which are earth oriented and contain an on-board star tracker system. Estimates are given for the horizon sensor development effort based on a mid-to-late-1965 go-ahead.

2.5.5 Data Parameters - The information obtained from this experiment constitutes an in-orbit evaluation of the performance of a horizon sensor. To

TABLE 2.5-2
HORIZON SENSOR ACCURACY EXPERIMENT PHYSICAL PARAMETERS

ITEM	VOLUME		WEIGHT	AVERAGE POWER (WATTS)	
	CU.FT.	L – W – H (INCHES)	(POUNDS)	PEAK	NOMINAL
Experimental Devices Horizon Sensor (Modified Gemini)	.18	7 x 5 x 4 5 x 5 x 5	12	12	10
Support Equipment Programmer Electronics Signal Conditioning Electronics Power Supply and Filter Recorder	 .03 .06 .1 .3	 5 x 3 x 3 5 x 5 x 4 10 x 8 x 2 14 x 9 x 5	 5 7 15 15	 10 10 10 40	 10 10 10 40
Subtotal	.67		54	82	80
Master Reference* (less gyro) 1. Star Tracker Head 2. Star Tracker Electronics	 .87 .51	 17 x 11 x 8 16 x 11 x 5	 23.5 21.6	20	15.4
Total	2.05		99.1	102	95.4

*Required if not available on carrier vehicle

obtain this data requires a precision measurement of the vehicle attitude as well as horizon sensor outputs. As previously discussed in Paragraph 2.5.3, "Test Method", precision ground tracking, star tracking, and gyro angles as a function of time are required. Reduction of the star tracker and gyro angle data requires an "estimate" of vehicle attitude as determined fron the on-board control system outputs.

Table 2.5-3 is a tabulation of the experiment parameters to be telemetered for this test. The priority or relative need for the signal is indicated by the order in which the parameter is listed. Real-time parameters are available upon

TABLE 2.5-3
HORIZON SENSOR ACCURACY EXPERIMENT DATA PARAMETERS

DATA POINT	SIGNAL FORMAT	RANGE OF PARAMETER	SAMPLE RATE (FREQ. RESP.)	ACCURACY	RECORD	ESSEN-TIAL	DESIR-ABLE	CON-TINUOUS	REAL TIME
Horizon Sensor									
1. Pitch Output	Digital	12 Bits	20 time/sec.	± 1 Bit	X	X		X	
2. Roll Output	Digital	12 Bits	20 time/sec.	± 1 Bit	X	X		X	
3. Loss-of-Track	Digital	On/Off			X	X			
4. Input Voltage	Analog	28 VDC		± 5%	X	X			
Master Reference									
4. Star Tracker Gimbal No. 1	Digital	17 Bits	20 time/sec.	± 1 Bit	X	X		X	
5. Star Tracker Gimbal No. 2	Digital	17 Bits	20 time/sec.	± 1 Bit	X	X		X	
6. Input Power	Analog	28 VDC		± 5%	X	X			
Miscellaneous									
7. Time Reference	Digital	22 Bits	20 time/sec.	± 1 Bit	X	X		X	
8. Horizon Sensor Head Temperature	Analog	0 – 160°F		± 5%	X		X		
9. Horizon Sensor Electronics Temperature	Analog	0 – 160°F		± 5%	X		X		
10. Star Tracker Temperature	Analog	0 – 160°F		± 5%	X		X		
11. Horizon Sensor Pre-Amp.	Analog		200 cps	± 5%					X
12. Horizon Sensor Pos. Det. Output	Analog		30 cps	± 5%					X
13. Star Tracker A.G.C.	Analog			± 5%					X
14. Star Tracker Gimbal Torquer	Analog			± 5%					X
15. Star Tracker Track Indication	Digital	On/Off							X
Required Spacecraft Signals									
1. Horizon Sensor Roll	Analog	0–5 V	5 cps	± 2%	X	X		X	
2. Horizon Sensor Pitch	Analog	0–5 V	5 cps	± 2%	X	X		X	
3. Horizon Sensor Loss-of-Track	Digital	On/Off			X	X		X	
4. Attitude Gyro, Pitch	Analog	360°	5 cps	± 1%	X	X		X	
5. Attitude Gyro, Roll	Analog	–80° to +80°	5 cps	± 1%	X	X		X	
6. Attitude Gyro, Yaw	Analog	–5° to +5°	5 cps	± 2%	X	X		X	
7. Rate Gyro, Pitch	Analog	± 2°/sec.	5 cps	± 2%	X		X	X	
8. Rate Gyro, Roll	Analog	± 2°/sec.	5 cps	± 2%	X		X	X	
9. Rate Gyro, Yaw	Analog	± 2°/sec.	5 cps	± 2%	X		X	X	
10. Spacecraft Main Power Buss	Analog	20 to 30 VDC	60 cps	± 1%	X		X	X	

ground command for detailed system analysis. Additional parameters may be added to the list as desired.

2.5.6 Vehicle Orbit and Attitude Requirements - Table 2.5-4 contains a summary tabulation of the most desirable orbit and vehicle stabilization requirements. A discussion of the effect of variation of the parameters is contained in the following paragraphs.

Altitude - The orbital altitude for conducting the experiment is not critical with 100 to 900 nautical miles being acceptable. It is essential that the planned orbit altitude be known so that the horizon sensor can be evaluated and optimized during ground tests for that altitude. Horizon sensor scale factors, cross coupling and to some extent the null vary as a function of altitude.

Eccentricity - The horizon sensor output is affected by the eccentricity of the orbit. The cross coupling of the pitch into roll and roll into pitch is a function of the cotangent of one-half of the subtended angle of the earth as seen from orbit. In other words, the cross coupling will vary as altitude varies. In the data reduction, the eccentricity factor will have to be taken into account. Eccentric orbits introduce other errors as well, but for an apogee of up to 1000 miles and perigee of 100 miles the other errors are small.

The most desirable orbit from the data evaluation viewpoint, as well as from the viewpoint of optimizing a simple design, would be a low altitude orbit with an apogee to perigee altitude ratio of not over 2 to 1.

Inclination - In order to obtain a maximum amount of possible variation in the IR gradient due to seasonal variation a near polar orbit is desired. Inclination greater than 70° will give summer-winter data on each orbit. A 20° - 30° inclination would provide only a small amount of the possible seasonal variation within the field of view of the sensor in the two week period recommended for this experiment.

Vehicle Stabilization - The vehicle must be earth oriented and stabilized in three axes. If the vehicle utilizes horizon sensors for establishing local vertical and either a strap-down gyro system or a platform, the vehicle alignment can be expected to be better than 1° in pitch and roll and 5° in yaw. Operating the vehicle with a small deadband (less than 2°) and with low rates (less than 0.1°/

TABLE 2.5-4

HORIZON SENSOR ACCURACY EXPERIMENT
ATTITUDE AND ORBIT REQUIREMENTS

PARAMETER	DESIRED CONDITION
a) Altitude	100 to 900 Nautical Miles
b) Eccentricity	Less than 0.102
c) Orbit Inclination	Polar (greater than 70°)
d) Stabilization	Earth Oriented in three axis Vehicle control deadband less than 2° Vehicle rates less than .1°/sec. Alignment to orbit plane – better than 5° in yaw.

sec) will provide a sufficiently stable "platform" for this experiment. Increasing the deadband or rates or decreasing the alignment accuracy will degrade the accuracy of the experiment. Ideally, the horizon sensor would be operated at null for the experiment. However, this may impose a severe constraint on the vehicle.

2.5.7 Equipment Support and Data Handling Requirements - In order to perform the experiment, support equipment on the vehicle is required. The signal conditioning, programming, master reference and environmental control requirements will place certain restrictions and constraint on vehicle selection and design. The requirements for equipment support and data handling are discussed in the following paragraphs.

Signal Conditioning - In order to obtain the required accuracy and to minimize the transmission noise problems, as much data as possible should be transmitted in digital format. This will require, in some cases, the addition of analog to digital converters to either the experimental package or the signal processing electronics. Figure 2.5-2 illustrates one method which could be utilized for providing serial bit words for recording. The horizon sensor, master

HORIZON SENSOR ACCURACY EXPERIMENT
BIT DATA PROCESSING BLOCK DIAGRAM

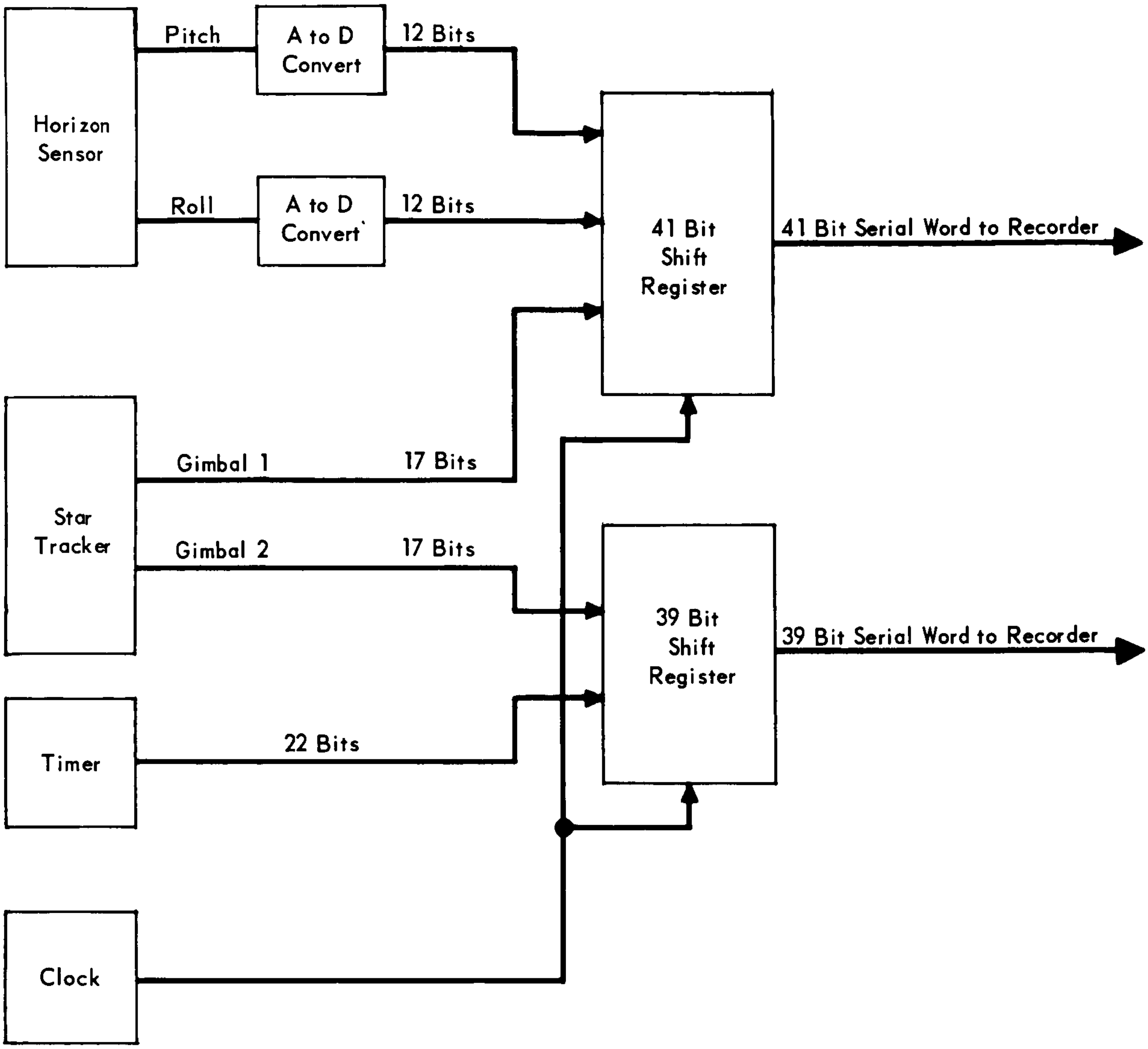


FIGURE 2.5–2

reference and time reference signals are processed through shift registers which are operating at a rate compatible with recording capability and the required sampling rate.

Programmer - This unit is of special design to serve the purposes required by this experiment. It will consist of the necessary electronics, relays, memory circuits, time reference and logic networks to:

a. Point the star tracker to the required position and initiate a search to establish a star lock-on condition.

b. After star lock-on slave the two-axis gyro to the star tracker.

c. Break lock and repeat (a) for a new star.

d. After lock on, begin data storage.

e. When star tracker gimbals reach a predetermined position, break lock-on and resequence the star tracker search/lock-on procedure.

f. Turn off recorder during star tracker search periods.

g. Update time reference system upon ground command.

h. Revise star tracker search angles upon ground command.

i. Sequence data transmission to ground stations at predetermined times or upon ground command.

j. Turn systems on and off either upon ground command or in a preprogrammed time sequence.

k. Sense failures of experimental package which would effect performance of the primary vehicle and turn the experiment packages off as required.

l. Provide a precision time reference.

Power Conversion and Filters - The power conversion and filter package provides the necessary RFI filtering and power conversion (DC to AC, DC to DC and voltage regulation) required by the individual experiment systems. The major power requirement is for 400 cycle, 26 volt AC.

Recorder - In order to obtain the maximum amount of usable data, and in order to obtain data from areas where ground tracking stations are not available, an on-board recorder is required. Through use of ground command signals, the recorded data is transmitted in compressed time. Because of the limited frequency response characteristics of space qualified recorders only a few carefully selected

HORIZON SENSOR ACCURACY EXPERIMENT FIELD OF VIEW REQUIREMENTS

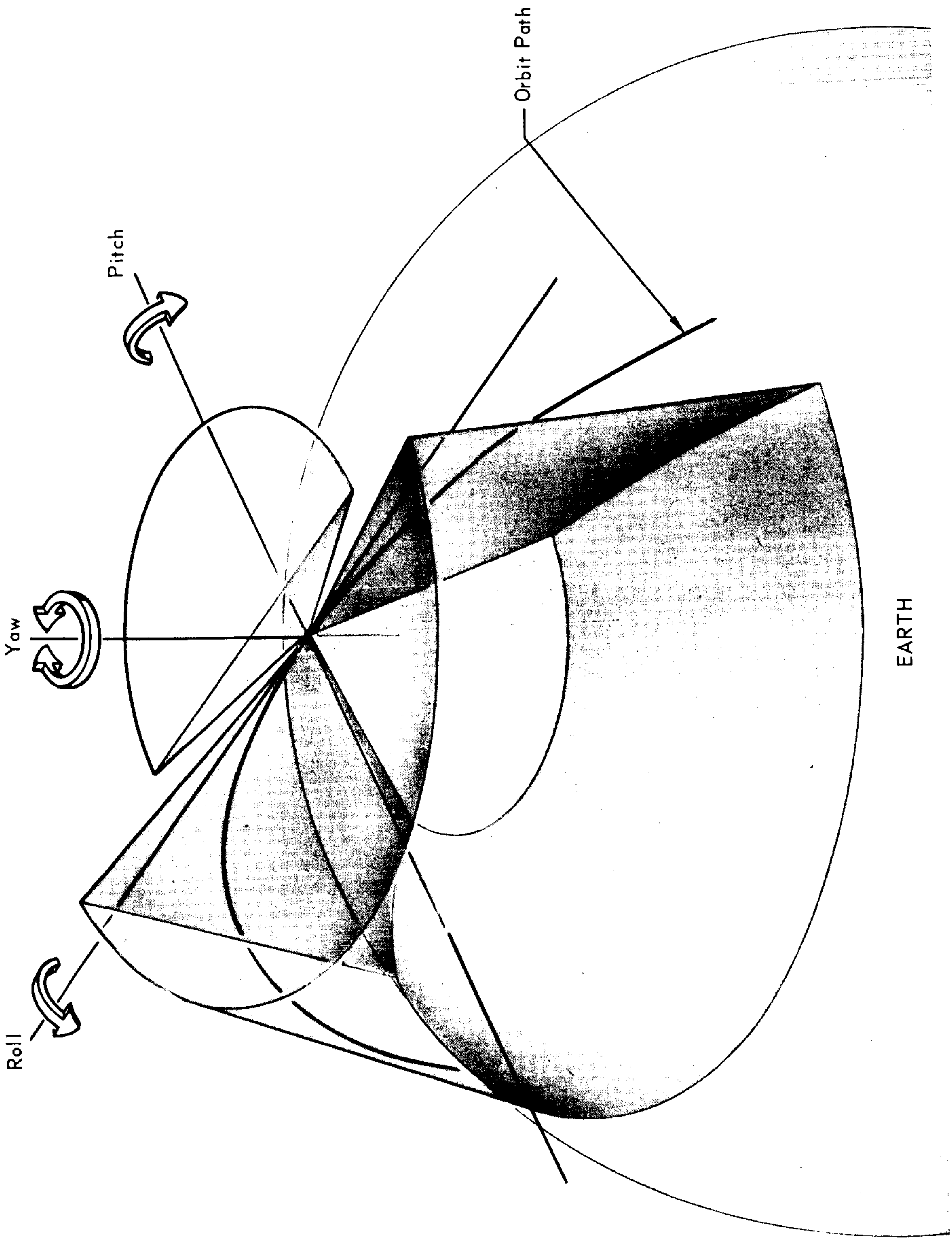


FIGURE 2.5-3

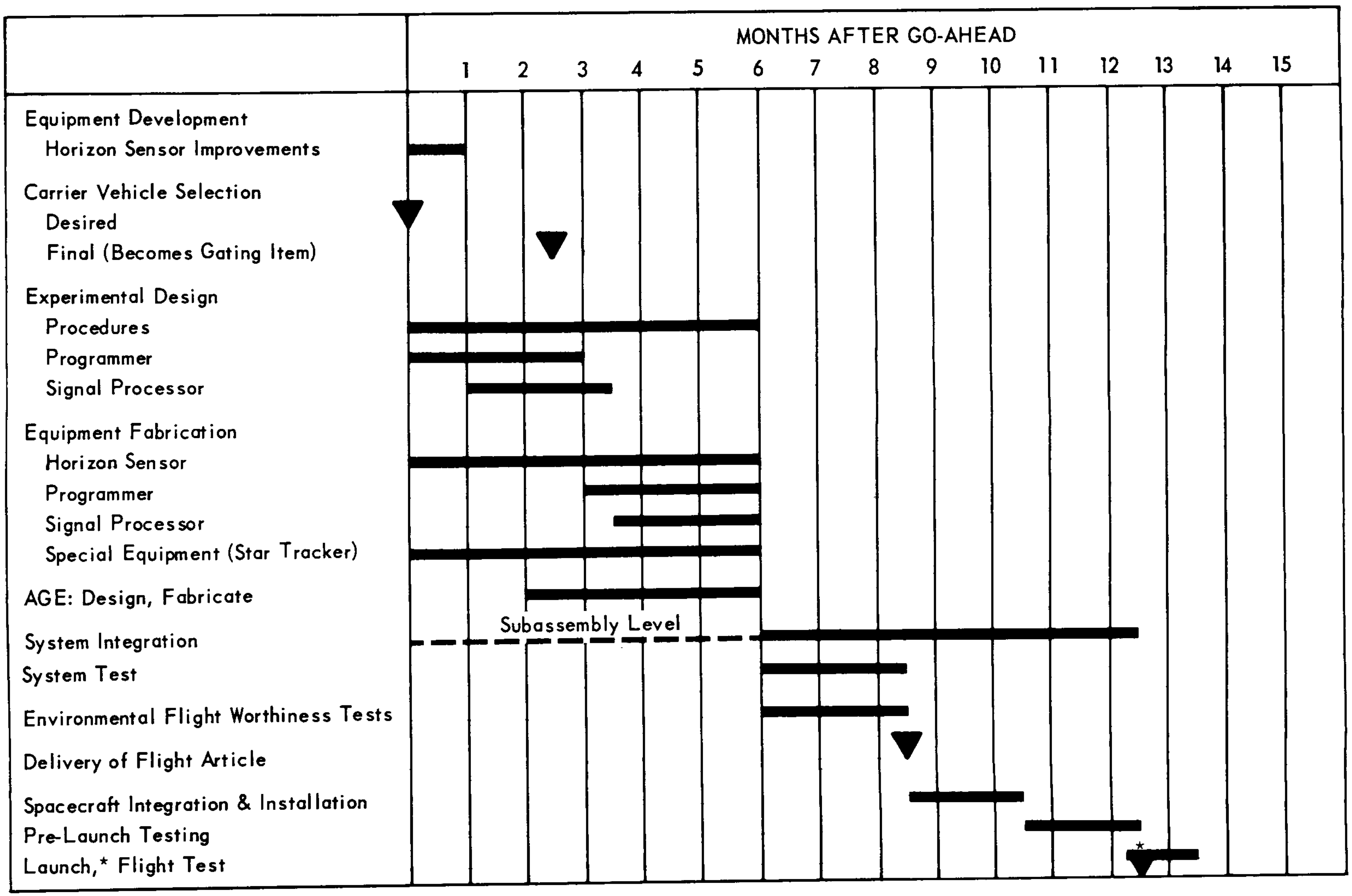
parameters will be taped. Additional operational parameters will be available during real-time transmission in order that over-all systems operation may be more carefully evaluated. The technique of recording as well as real time data transmission is utilized to assure that information obtained from the experiment will establish the accuracy of the sensor and provide data on which to establish the design changes necessary to improve the horizon sensor design.

Master Reference - The required master reference is a star tracker and a two-axis gyro. Since it is not expected that the carrier vehicle will normally contain a star sensor, the recommended master reference system to be added is an OAO star tracker. The star tracker and gyro, as applied, are discussed in greater detail in Appendix A, "Master Attitude Reference System Considerations".

Environmental Control - The horizon sensor head is located essentially external to the carrier vehicle's external mold line. In order to maintain the head temperature below 130°F a passive paint finish is used. To maintain the head temperature above 0°F, heater elements are installed in the units. The electronics package is presently designed for cold-plate operation (40°F to 130°F). Prior to installation and flight, a thorough analysis of the vehicle's thermal characteristics will be required to determine the need for additional heaters and/or cold plates.

Mounting - It is necessary that the horizon sensor and the master reference be very precisely aligned with respect to each other and with respect to the vehicle rotational axes. Alignment to 10 arc seconds in the pitch and roll axes and 6 arc minutes in the yaw axis is desired. In addition, the star tracker and the horizon sensor head require large unobstructed fields-of-view. Figure 2.5-3 is a sketch representing the field-of-view requirements. The OAO star tracker field-of-view is a cone of 150° with the central axis of the cone parallel to

HORIZON SENSOR ACCURACY EXPERIMENT DEVELOPMENT PLAN



*Depends on carrier vehicle – may vary from 1 month to 8 months after installation.

FIGURE 2.5–4

the yaw axis (only one-half of the cone is shown in the figure for clarity). The horizon sensor field-of-view is from the side of the vehicle with a view of $\pm 80^{\circ}$ normal to the longitudinal axis and $+ 15^{\circ}$, $- 60^{\circ}$ in the vertical.

Electrical Energy Requirements - The total energy requirement for the proposed two week mission with on-off cycles as defined in Paragraph 2.5.8 is 4,400 watt-hours not including the star tracker. The star tracker system requirements for the same time span is 2,400 watt-hours.

Ground Commands - Several commands are necessary through the ground link to the experiment and are summarized below:

a. Update the time reference system at least once each eight hour period in orbit.

b. Change program or sequence.

c. Provide manual override to obtain telemetered test points for failure analysis.

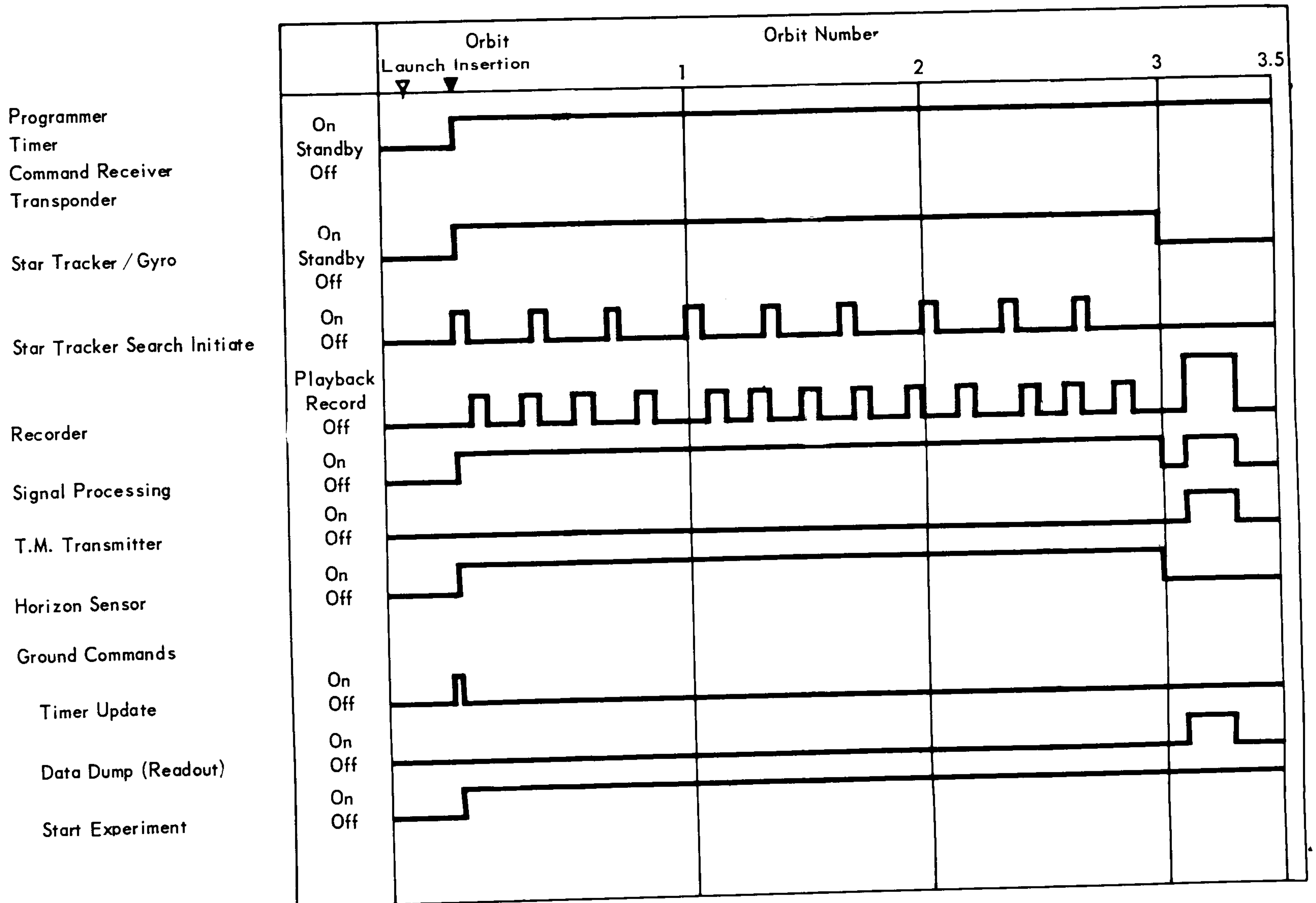
d. Turn experiment on, off, or place in standby condition.

e. Command data dump (data retrieval from recorder).

Data Handling - Appendix B contains a discussion of the general problems associated with the overall data handling system. Initiation of the design of the data handling subsystem of the experiment should start as early in the program as possible. The experiment is complex and requires a large number of channels to provide sufficient data to be entirely useful. Many of the signal conditioning requirements should be incorporated in the equipment to provide compatibility with the telemetry equipment.

2.5.8 Experiment Flight Test Plan - The overall flight test plan for planning purposes is shown in the flow diagram of Figure 2.5-4. The schedule is based on best estimates of the required development, systems integration and

FLIGHT TEST SEQUENCE, HORIZON SENSOR ACCURACY EXPERIMENT



Note: See Figure 2.5–6 for over-all mission profile.

FIGURE 2.5–5

testing and test time. Horizon sensors have a history of being highly susceptible to RFI; therefore, RFI problems may cause a lengthening of the systems integration phase. The schedule is predicated on an early selection of the carrier vehicle in order that package design and interference specifications may be firmed-up prior to the start of manufacturing.

An early trade-off between ground commands and stored commands should be made. As the programmer stored commands for star tracker pointing are reduced in quantity, the number of ground stations and commands for the same control increases. If the ground stations are available, the size and weight reduction may be worthwhile.

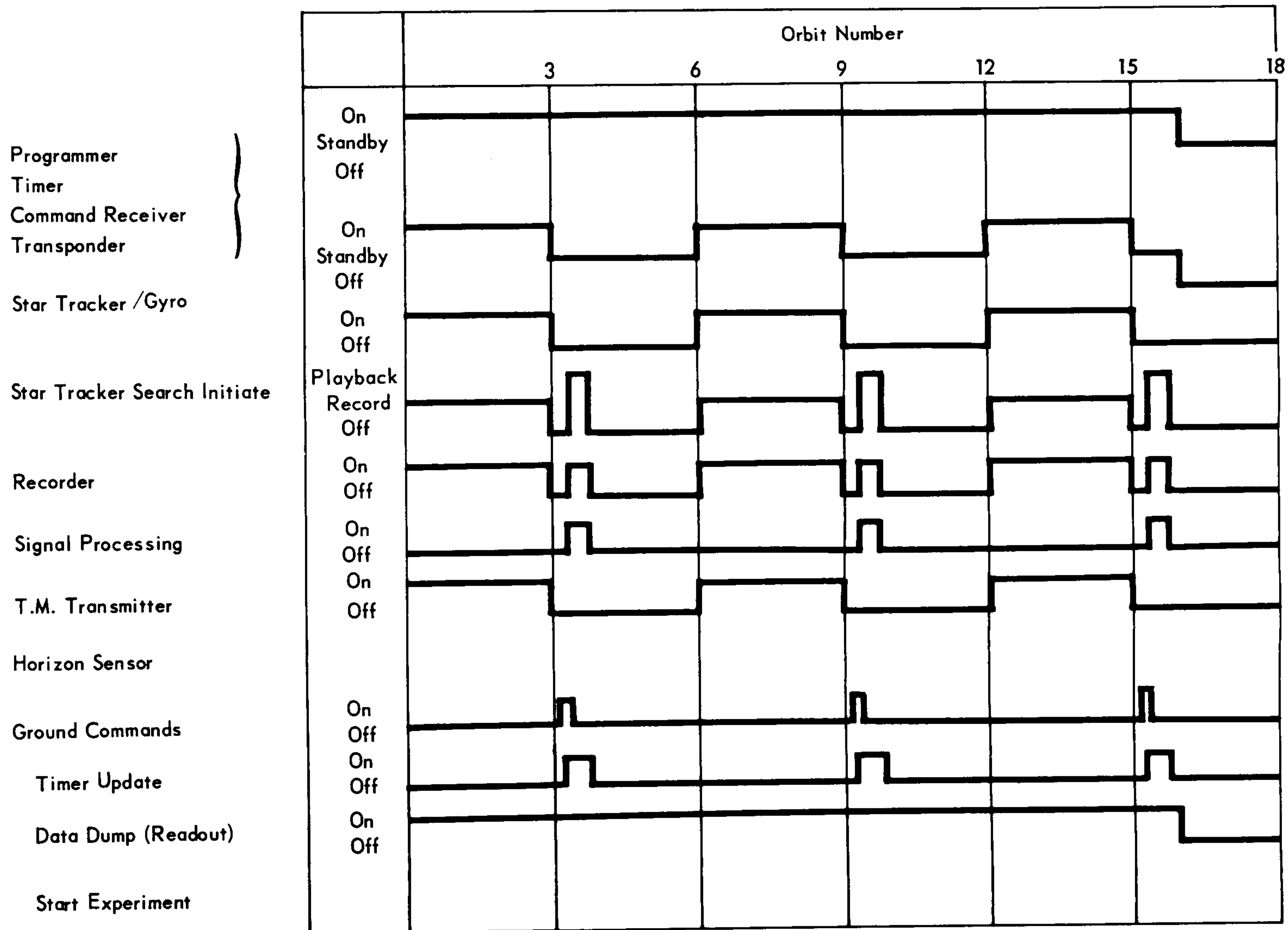
Figures 2.5-5 and 2.5-6 present a time plot of the operation sequence. Figure 2.5-5 illustrates the individual functions in small time increments, whereas Figure 2.5-6 illustrates the long term, several orbit functions or operating periods. The experiment is not operated continuously in orbit. The units are sequenced on and off during some orbits and left in an off condition for other orbits. A recommended two-week test sequence, which depends upon the orbit inclination, period and tracking stations, is as follows:

a. After orbital insertion and vehicle stabilization the experiment package is turned on.

b. The star tracker is energized and commanded to a predetermined angular position. The gimbal positions are based on the predicted position of a trackable navigation star as a function of the orbit path, vehicle orbital position, season and launch time.

c. The star tracker acquires and tracks a star; the two axis memory gyro is slewed to the tracker position. Star tracker lock-on is broken and the

MISSION CYCLES, HORIZON SENSOR ACCURACY EXPERIMENT



Note: Repeat above cycle during orbits 65 through 81, 131 through 147 and 197 through 213.

FIGURE 2.5-6

tracker begins search for a new star. After the second lock-on the data recorder is turned on for five minutes.

d. The recorder is turned off for 15 minutes (standby) and then turned on for 5 minutes.

e. Step (d) is repeated until star lock-on is lost and then steps (b) through (e) are repeated.

f. The cyclic on-off sequence is continued for three orbits.

g. The experiment is placed in stand-by or off condition for 3 orbits.

h. Repeat steps (b) through (g) three times.

i. The experiment shall be turned off for three days and then sequence (b) through (h) repeated, starting on the fifth day.

j. Repeat sequence (b) through (h) starting on the ninth and thirteenth day in orbit.

This sequence will provide at least 624 minutes of data, assuming the star tracker has lock-on at least 80% of the time. This will provide a nominal amount of data for a realistic evaluation of the statistical variation of the horizon sensor performance about its nominal. Information on seasonal (northern and southern hemispheres), day-night and tide-effect variations is obtained over a wide range of latitudes and longitudes. Lengthening the test or increasing the recording time will provide additional data for statistical analysis.

Post Experiment Data Reduction - Figure 2.5-7 is a line flow diagram of the data reduction process. The steps are listed below:

a. Ascertain, through use of the on-board horizon sensor and gyro system, the approximate attitude of the vehicle with respect to earth local vertical as a function of time.

HORIZON SENSOR ACCURACY EXPERIMENT
DATA REDUCTION FLOW DIAGRAM

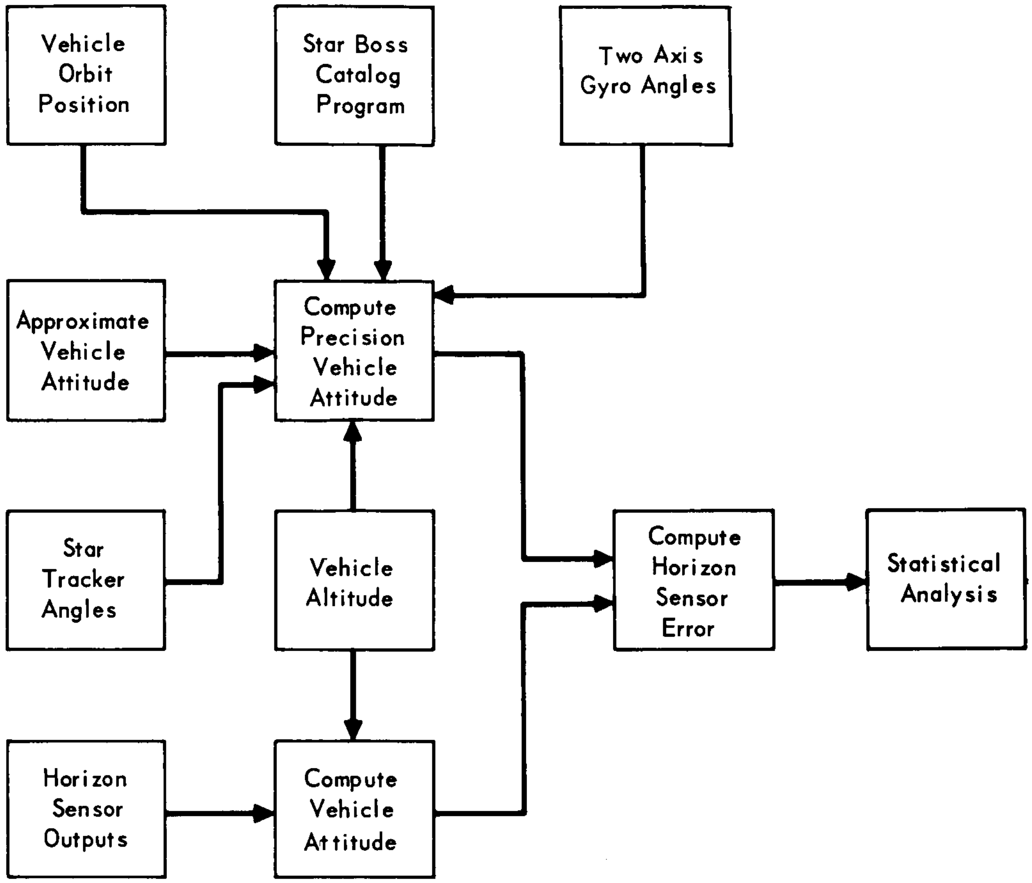


FIGURE 2.5–7

b. Time correlate the vehicle orbit position with the data from (a) above.

c. Through use of vehicle orbit position, approximate attitude, Boss General Star Catalogue and the star tracker and gyro angles during track, determine which specific stars are being tracked.

d. Update the vehicle attitude information using (c) above to derive precision data on vehicle attitude. Use of a single star tracker and a gyro provides sufficient data to compute the vehicle horizontal plane position with respect to the local horizontal coordinates.

e. Compute the horizontal plane position as established by the horizon sensor outputs as a function of time.

f. Compare the data of (d) and (e) above to obtain a time history of horizon sensor error as a function of time and vehicle attitude.

g. Perform a statistical analysis to establish a measure of horizon sensor performance as a statistical variation about some nominal.

Upon completion of the test and subsequent analysis of the results, it may be desired to modify the horizon sensor and retest. Several flights may be required to reach an optimum design which provides high accuracy with only small variations due to the atmospheric anomalies. The first flight will provide a measure of the performance of an existing design. Pre-flight ground tests can be so designed that a measure of the effect of theoretical variations to be expected in orbit can be established. The flight test will determine the sensitivity of the unit to actual anomaly variations.

2.6 Gas Bearing Performance Tests

2.6.1 Objectives - The objectives of this experiment are to prove the stability and to determine run-up and run-down characteristics of gas bearings at various ambient gas pressures in a zero-g environment. The results of this test are applicable to the development of long-life bearings for inertial equipment intended for space use.

2.6.2 Background - Demands for increased reliability in guidance and control systems have incurred equivalent demands on the development of gyroscopic instruments. An increase in this reliability depends largely on improving the operating life of the high speed motor bearings. In recent years, manufacturers have begun to investigate gas bearings as a possible solution to the problem. The principles of gas bearings have been known and demonstrated during the past fifty years. But it has only been recently that they have received serious consideration for instrument applications.

There are two categories of gas bearings; hydrostatic and hydrodynamic, each of which contain a multitude of possible designs. Both types support the journal on a cushion of air. The hydrostatic bearing receives its gas supply externally, either from an expendable source or by pumping and recirculating the available gas. The size and power requirements of the supply system limit the use of hydrostatic bearings in space. The hydrodynamic bearing generates its own gas cushion by the dynamic action between the journal and the bearing housing. Hydrodynamic bearings never achieve a completely stable condition because of the minute unbalances in the rotor. Each bearing design has a critical speed at which it will become unstable. If the journal is rotating at a speed below the critical speed for its configuration, the locus will converge to a circle or ellipse and rotate at a frequency approximately equal to the spin motor rotational frequency. This is known as "synchronous whirl". However, if the rotating speed reaches or passes critical, the locus frequency will rapidly drop to approximately half that of the spin motor and the locus will rapidly diverge. This is known as the "half-frequency whirl" instability (Reference 1). Occurrence of this phenomena is basically a function of bearing stiffness, bearing load, and wheel speed. The stiffness of hydrodynamic bearings is a function of several variables including air gap, ambient gas pressure, and journal speed. Although a higher ambient gas pressure increases the bearing stiffness, it also increases the power required to overcome the additional windage losses in the system. Bearing loading varies as a function of the sine of the angle between the rotor spin axis and the gravity vector. The critical speed varies as a function of the bearing load and thus, will be quite low when the spin axis is parallel to the gravity vector.

Ground testing of plain, cylindrical journal air bearings with the spin axis vertically oriented was conducted by the General Electric Company in 1960 (Reference 2). General Electric was able to obtain marginally stable operation in only three of seventeen configurations. Later studies have shown that the plain, cylindrical journal gas bearing is the least stable configuration of the journal bearing. This is believed to be chiefly caused by the symmetry of the design. The pressure variation in the gas column around the journal is a sine function with a frequency approximately equal to rotor rotational frequency during stable operation. At the critical speed, the supporting air column apparently becomes resonant and fails.

Later investigation into the instability of journal bearings has lead to several innovations in gas-bearing design which are aimed at providing stability by preventing the sinusoidal air column. Design configurations include: slots in the housing face (Reference 3); non-circular housings; and movable pads. The first configuration contains a multitude of design variables including the number of slots, the relative width and depth of each slot, and direction of the slots (i.e., axial versus spiral). The slots introduce steps into the air-column pressure variation. Investigation of non-circular housings has been primarily concerned with elliptical housings of varying eccentricity and relative average clearance. Though the ellipticity permits a smooth pressure variation it is not purely sinusoidal. Movable pads as a means of achieving stability is receiving much attention at the present time. One design of particular interest is the pivoted pad bearing which consists of many small pads, each free to move in two axes. These allow the bearing to simultaneously tilt relative to lines perpendicular and parallel to the journal axis. The break in the smooth housing wall prevents the sinusoidal pressure variation.

There have been two alternate approaches proposed for orbitally testing gas bearings. One method is based on the use of small gyroscopes with gas bearings supporting the spin axis. Such a test could provide a qualitative yes-no answer on whether that configuration was stable up to its design speed, but would provide no data on the cause and effects of instability if it should occur. The second and preferred method consists of testing a fully instrumented gas bearing supporting a motor driven inertia wheel. This method will concentrate on monitoring the gas bearing stability and proving the capability of gas bearings to function in the zero-g environment.

Ground testing one of these devices with the spin-axis oriented vertically will result in little or no load on the journal bearings, similar to conditions expected in the space environment. To operate in the ground environment, the test device must be designed with thrust bearings of a size which will support the weight of the rotor assembly against the one-g gravity vector. In orbit, thrust bearings are required only to maintain the rotor assembly in its proper location

GAS BEARING EXPERIMENT
ELECTRIC MOTOR DRIVEN ROTOR

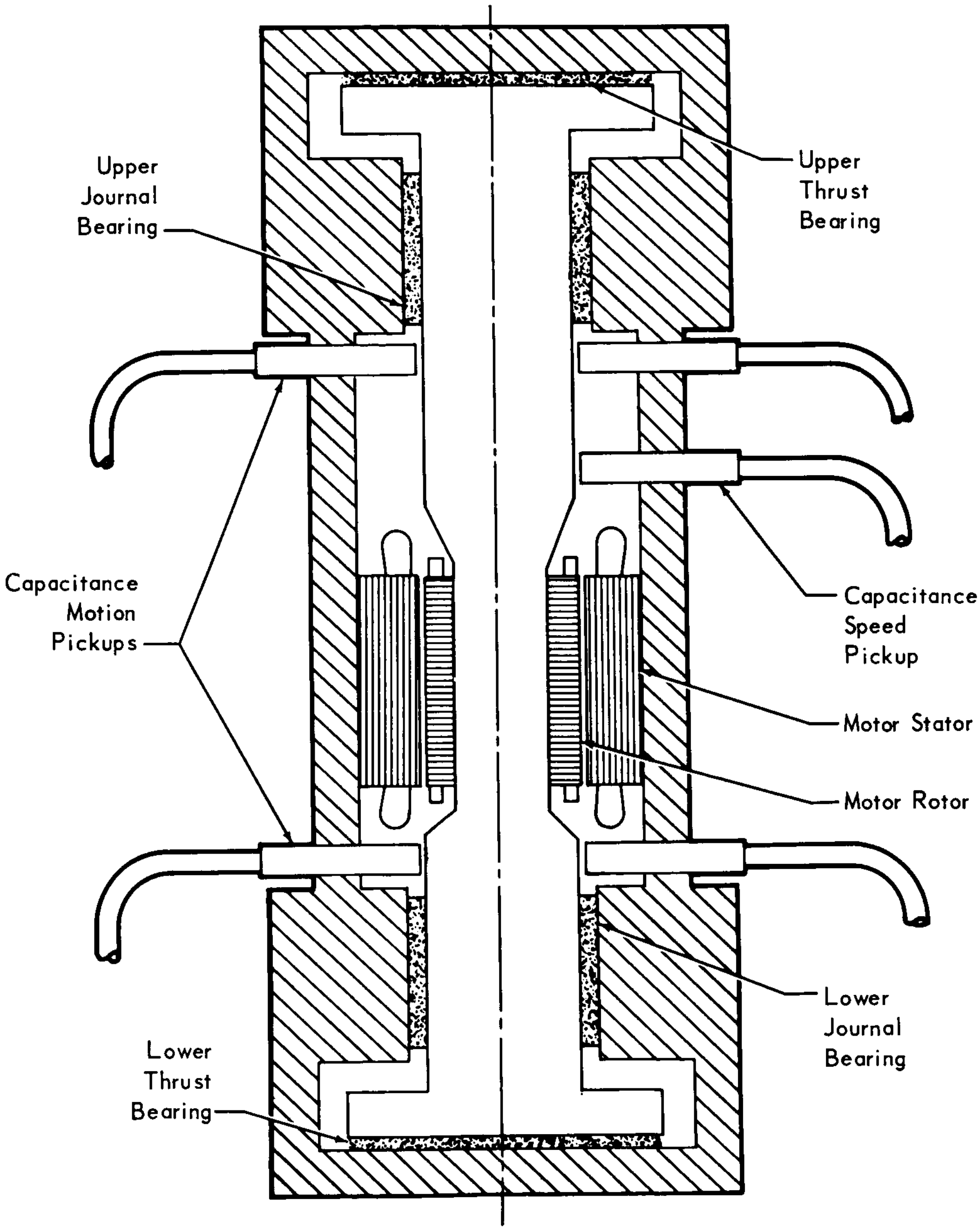


FIGURE 2.6-1

when it is subjected to extraneous axial accelerations. The possible stabilizing influence of this thrust bearing makes it necessary to conduct final proof testing in a zero-g environment. Figure 2.6-1 pictures a typical rotor and gas bearing configuration which is capable of operating in the vertical mode.

2.6.3 Functional Description - The experimental system consists of the components shown in the Functional Block Diagram (Figure 2.6-2). The programmer signal conditioner and power supply are considered experimental support equipment and are described in paragraph 2.6.7. The experimental equipment and the test method are discussed below. Based on the three journal bearing designs presently under development, the experiment proposes to test three rotor assemblies which are identical except for bearing design. Each assembly has a variable speed motor with a full speed momentum of approximately 0.5 million cgs units. The journal bearings for the three units are tilted pad, elliptical, and slotted. The shafts are instrumented with four capacitive pickoffs at each end, located at 90 degree intervals around the shaft. A single capacitive pickoff is located on each shaft to monitor rotor speed by generating a pulse each time a specific spot on the shaft passes it. Each case is fitted with a valve which will be capable of venting the atmosphere within the case. The valve operation is controlled by the programmer and a pressure transducer.

B. Test Method - Each rotor assembly is subjected to three tests. Stability is determined by applying voltage to the rotor in steps until full voltage is reached. At each step, the rotor speed is measured. From full speed, the voltage is reduced in a similar fashion until the rotor is stopped. This test will either prove stability throughout the speed range or determine the critical speed. The stability tests will then be repeated under normal starting conditions by applying the full voltage to the rotor and monitoring the experiment until full speed of

GAS BEARING EXPERIMENT FUNCTIONAL BLOCK DIAGRAM

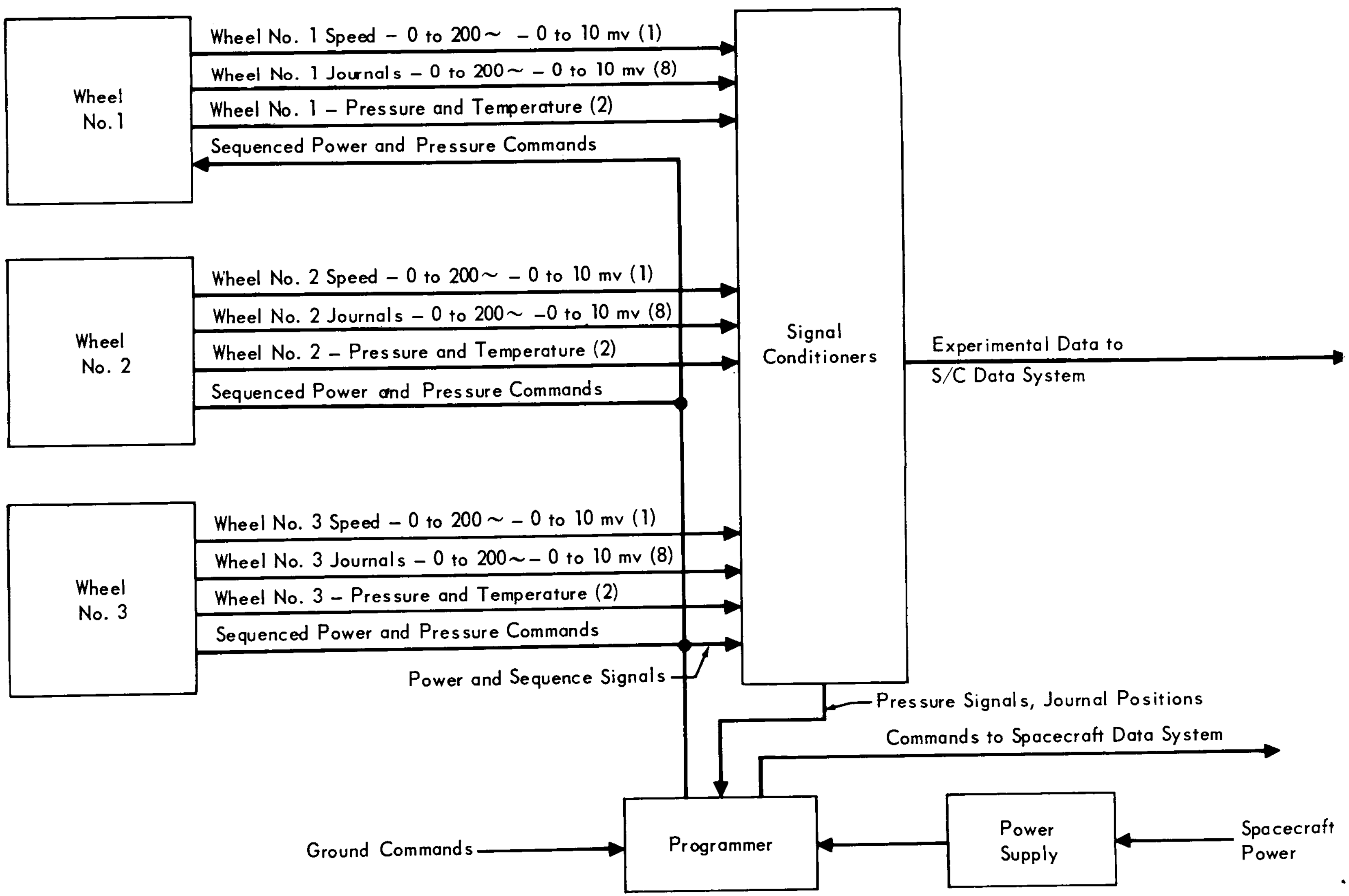


FIGURE 2.6-2

24,000 rpm is reached. The data from this test will also provide speed as a function of time which defines the motor starting characteristic. If instability should occur during either of these tests, power is automatically removed from the motor by an instability detector. The run-down characteristics are measured by monitoring wheel speed after power is removed from the rotor when it is running at full speed. This test is designed to provide a measure of windage losses in the system. The first two tests are conducted on each rotor assembly in sequence. The third test can be conducted on any rotor simultaneously with any tests on the other rotor assemblies. One test series will consist of the three tests being completed on each unit. Five series of tests will be scheduled. The only variable which is introduced is the case pressure. Between each test series, the pressure may be changed upon ground command. Control of the pressure both upward and downward is possible using a pressure tank and regulator, and providing a ground command for each distinct test pressure used. A simpler procedure is to vent the pressure downward only using one ground command. Caution must be exercised with the second method to avoid a pressure reduction before all tests have been completed. The test pressure levels will be determined by the speed and type of bearing.

2.6.4 Experiment Physical Parameters - The physical and operating parameters of the recommended equipment is provided in Table 2.6-1. The table contains a summary of the size, weight, and power requirements of the basic experiment package. For the equipment requiring development effort, the values given are estimates.

2.6.5 Data Parameters - The information obtained from this experiment is an in-orbit evaluation of gas-bearing stability. To obtain this data requires measurement of the experiment parameters shown in Table 2.6-2. Priority 1 parameters

TABLE 2.6-1

GAS BEARING EXPERIMENT PHYSICAL PARAMETERS

EQUIPMENT	SIZE		WEIGHT (LBS.)	POWER	
	VOLUME ($FT.^3$)	L – W – H (IN.)		PEAK (WATTS)	NOMINAL (WATTS)
Experiment					
Wheel No. 1	0.065	7 x 4 x 4	6.5	7*	
Wheel No. 2	0.065	7 x 4 x 4	6.5	7*	
Wheel No. 3	0.065	7 x 4 x 4	6.5	7*	
Support					
Sequence, Logic, and Time Ref.	0.032	4 x 4 x 3½	1.6	3*	1*
Signal Conditioner	0.004	7 x 1 x 1	0.2	5*	2*
Power Supply	0.024	4 x 3 x 3½	2.0	21	4
Total	0.255		23.3	21	4

*Provided by Power Supply

are considered essential. These will show only that the wheel is stable, but not what it does. Priorities 2 and 3 would complement the Priority 1 parameters to complete the instrumentation of wheel operation. The desirable parameters (Priorities 4, 5 and 6) serve to provide a complete picture of the experiment operation. In addition to the spacecraft parameters, spacecraft angular rates must be monitored to determine load placed on the gas-bearing by vehicle motion (see Paragraph 2.6.6).

2.6.6 Spacecraft Control and Orbit Requirements - The nature of these experiments places very few restraints upon the spacecraft and none upon its orbit. During periods when the experiment is being performed, the vehicle angular rate and angular acceleration (combined with the distance from the vehicle c.g.) must not

TABLE 2.6-2
GAS BEARING EXPERIMENT DATA PARAMETERS

DATA POINT	PRIORITY	SIGNAL FORMAT AND FREQUENCY	PARAMETER RANGE	SAMPLE RATE (/SEC.)	ACCURACY	REMARKS
Wheel Speed – No. 1	1	Digital				
No. 2	1	0–400 pps	0–24,000 rpm		± 50 rpm	Monitored sequentially when power, applied to any wheel during rundown test, and upon ground command.
No. 3	1					
Wheel No. 1 – Journal No. 1	1					
No. 2	2					
No. 3	3					
No. 4	3					
Wheel No. 2 – Journal No. 1	1					
No. 2	2	Analog	0–1000 in.	2000	± 20 in.	Non-sinusoidal; monitored only when power applied to particular wheel or upon ground command. Frequency, amplitude and wave shape data required.
No. 3	3	0–400 cps				
No. 4	3					
Wheel No. 3 – Journal No. 1	1					
No. 2	2					
No. 3	3					
No. 4	3					
Wheel No. 1 – On/Off	1					Automatically monitored with ground override.
No. 2 – On/Off	1		On/Off			
No. 3 – On/Off	1					
Voltage Level – Reverse Start	4		–3VDC Nom.			
– Start	4		3 VDC Nom.			
– Step 1	4		5.5 VDC Nom.			
– Step 2	4		8 VDC Nom.			
– Step 3	4		10.5 VDC Nom.	1.0		Monitor only during stability test.
– Step 4	4		13 VDC Nom.		1 VDC	
– Step 5	4		15.5 VDC Nom.			
– Step 6	4		18 VDC Nom.			
– Step 7	4	Analog DC	20.5 VDC Nom.			
– Step 8	4		23 VDC Nom.			
– Step 9	4		25.5 VDC Nom.			
– Full	4		28 VDC Nom.			Monitor when any wheel power on.
Stability Test – On/Off	5					
Starting Test – On/Off	5		On/Off			
Re-Start Test – On/Off	5					
Run-down Test – On/Off	5					
Case Pressure – No. 1	1			0.1		Monitored throughout duration of test.
No. 2	1		6–15 psia		± 0.1psia	
No. 3	1					
Case Temperature – No. 1	6					
No. 2	6		0–120°F		± 2°F.	
No. 3	6					

produce more than 0.01 g's of acceleration on the experimental package. The equation which defines this acceleration is:

$$a \cong r\sqrt{\alpha^2 + \omega^4}$$

where:

a = acceleration on package

r = distance from c.m. of package to c.m. of vehicle

α = vehicle angular acceleration

ω = vehicle angular rate

For example, assuming that $\alpha = 0.1\ \omega$ and $r = 3$ ft., the vehicle can have an angular rate of up to 12.7°/sec. without exceeding the .01 g limitation. In addition to these, bearing loads will occur due to the gyroscopic torques introduced by the angular rate of the vehicle. The restrictions on vehicle motion prevent it from introducing appreciable loads into the bearings.

2.6.7 Experiment Support and Data Handling Requirements - The data handling requirements discussed in Appendix B are applicable. The transponder is not required since vehicle orbital position is of no importance to the experiment. A master reference is not required and no unique problems are expected in the areas of programming. Based on the flight test sequences described in Figure 2.6-4 and Figure 2.6-5, the five test cycles of this experiment will require 30 watt-hours of energy. Unique problems exist in the experimental support areas described below.

Signal Conditioning - The design of the Signal Conditioning unit requires special attention in two areas. The journal signals are analog AC signals which are distorted sine waves and vary in frequency from 0 to 400 cps. Direct demodulation for amplitude measurement is very difficult. In addition, the signal wave forms are important. High speed sampling may be employed to obtain the data. The wheel speed signals are digital pulses, varying from 0 to 400 pulses per second. A counter will measure the number of pulses in a given time period and transfer the count of an accumulator which will store the last count for transmission. Development of the experimental wheels will provide data concerning the shape and duration of this pulse. The need for a more accurate measure of speed will be investigated during the development phase.

Environmental Control - The test units are hermetically sealed and may be located in an unpressurized area. The temperature environment, however, must be kept in the region of 0 to +120 degrees Fahrenheit.

Mounting - The experimental package should be located as near to the vehicle, c.g., as possible to minimize the disturbance effect of the vehicle. The g-level limitation described in Paragraph 2.6.1 is directly proportional to this distance.

Commands per Experiment Sequence - The experimental package operates in conjunction with twelve possible ground commands. These are defined in Table 2.6-3. The twelve commands consist of one power on command, one automatic program initiate command, one data dump command, eight manual override commands and one power off command. In the event that the time reference fails, the automatic wheel speed monitoring can be replaced by command monitoring. Command monitoring of the journals offers a backup to the wheel speed measurement since the periodicity of the journal position is approximately equal to wheel speed when stable conditions exist.

2.6.8 Experiment Flight Test Plan - The development plan for the experiment is detailed in Figure 2.6-3. The gating factor in the estimated sixteen month time period from go ahead to installation is the development time required by the rotor assemblies. The use of three different bearing designs in this experiment was dictated by present gas-bearing technology. As the technology advances, one, or more, of these designs may prove to be more feasible. Therefore, the first step in the development of the rotor assemblies will be to select the quantity and type of bearings to be orbitally tested.

TABLE 2.6-3
GAS BEARING EXPERIMENT
GROUND COMMANDS

GROUND COMMAND	ACTION
Power On	Power is supplied to the experimental package. Programmer resets to zero.
Initiate Programmed Test	Programmer commences one automatic cycle
Data Dump	All recorded data is transmitted. Reset programmer
Lower Case Pressures	Programmed case pressure commands will be stepped to the next sequential position.
Wheel No. 1 Power On	Full operating voltage is applied to Wheel No. 1. Wheel No. 1 journals and all wheel speeds are monitored.
Wheel No. 2 Power On	Full operating voltage is applied to Wheel No. 2. Wheel No. 2 journals and all wheel speeds are monitored.
Wheel No. 3 Power On	Full operating voltage is applied to Wheel No. 3. Wheel No. 3 journals and all wheel speeds are monitored.
Monitor No. 1 Journals	No. 1 journals are monitored for a preset time.
Monitor No. 2 Journals	No. 2 journals are monitored for a preset time.
Monitor No. 3 Journals	No. 3 journals are monitored for a preset time.
Monitor Wheel Speeds	All wheel speeds are monitored sequentially and repeatedly for a preset time.
Power Off	Power is removed from the experimental package.

The automatic flight test sequence is defined in Figure 2.6-4. It commences automatically upon ground command. Each wheel is tested for stability throughout the speed range. Then a run-up and run-down test is performed on each wheel. The case pressure change which occurs at the end of the sequence is not automatic but requires a separate ground command. This makes it possible to repeat the automatic sequence without changing the pressure. The complete flight test plan tentatively uses five cycles of the automatic sequence as shown in the following list of steps.

GAS BEARING EXPERIMENT DEVELOPMENT PLAN

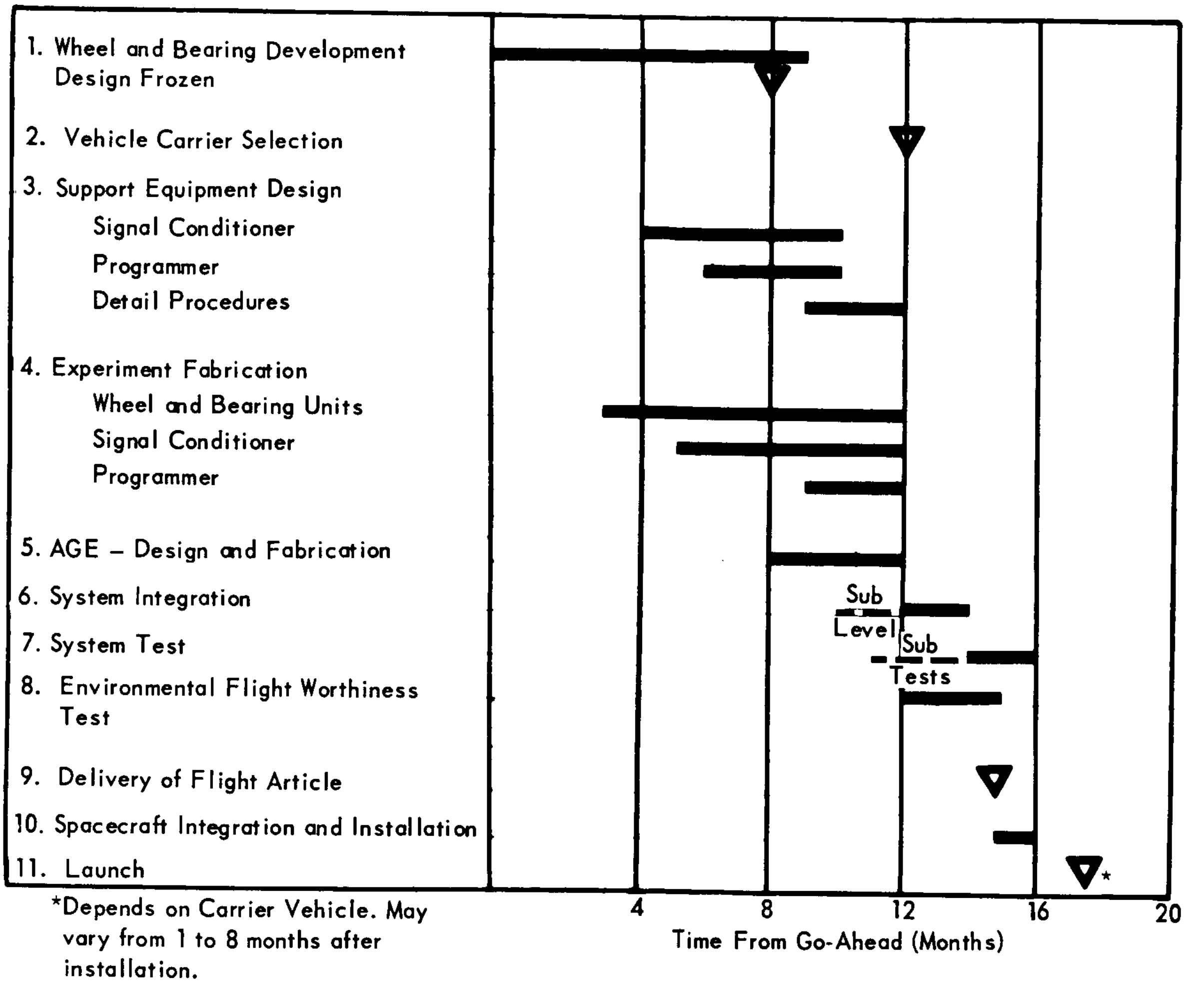


FIGURE 2.6–3

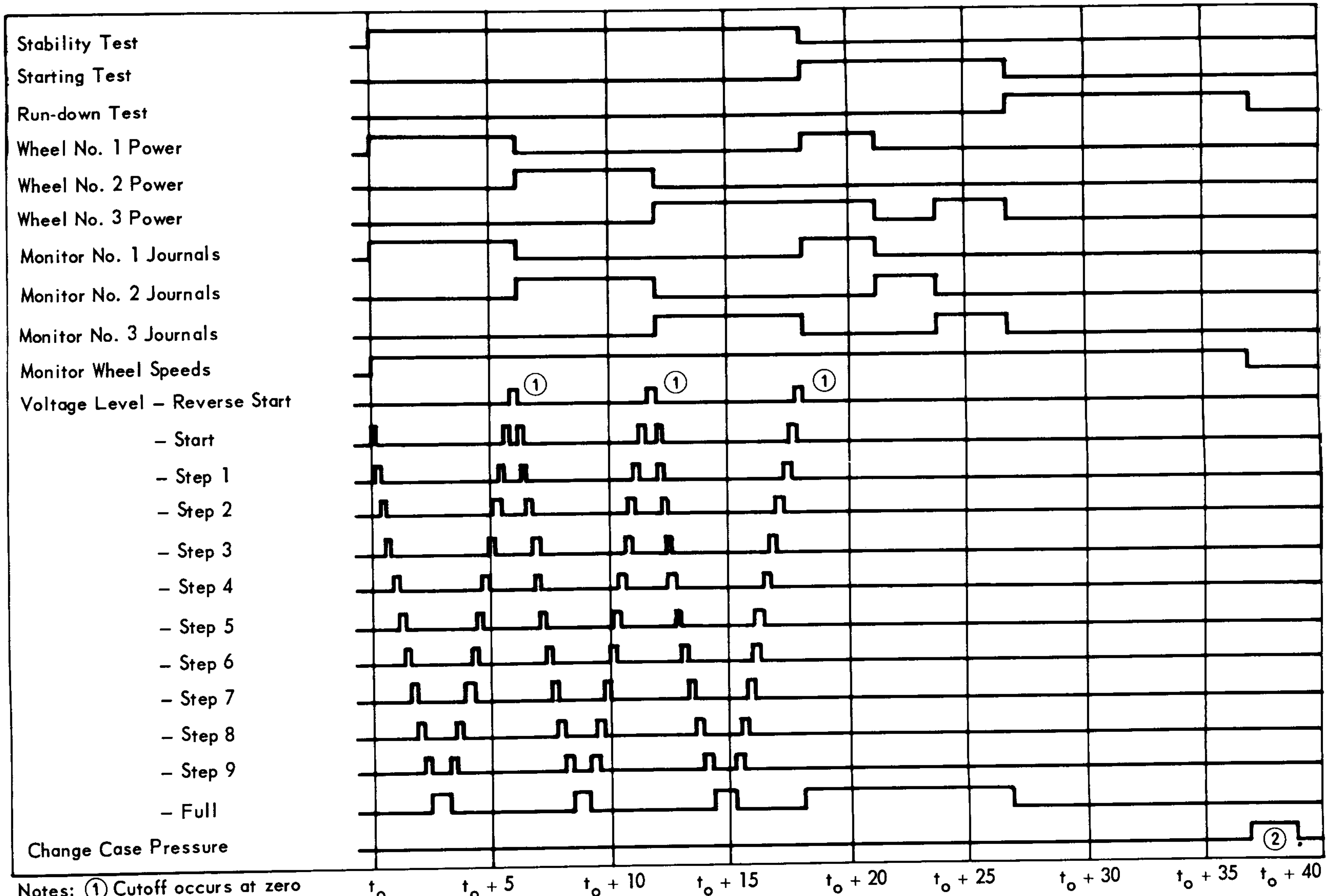
a. Apply starting voltage to Wheel No. 1 and establish the gas bearing.

b. Increase the voltage to Wheel No. 1 in ten steps. The final voltage step will result in full design speed. Journal position will be monitored throughout this operation. Any instability will automatically result in removal of power from the wheel.

c. Decrease the voltage to Wheel No. 1 in ten steps. The final voltage step will result in the starting voltage. The polarity of this voltage level

FIGURE 2.6-4

GAS BEARING EXPERIMENT AUTOMATIC FLIGHT TEST SEQUENCE



Notes: 1 Cutoff occurs at zero wheel speed
2 Pressure change occurs only upon ground command

will then be reversed for a sufficient length of time to stop the wheel's rotation. Journal position will be monitored throughout this operation. Any instability will result in removal of power from the wheel.

d. Repeat steps (a) through (c) using Wheel No. 2.

e. Repeat steps (a) through (c) using Wheel No. 3.

f. Apply full voltage to Wheel No. 1. The time required for the wheel to reach full speed will be determined. Journal position will be monitored throughout this step. Any instability will result in removal of power from the wheel.

g. When Wheel No. 1 has stabilized at full design speed, the motor excitation voltage will be removed. Wheel speed will be monitored and recorded during run-down.

h. Repeat steps (f) and (g) using Wheel No. 2.

i. Repeat steps (f) and (g) using Wheel No. 3.

j. When all wheels have completed run-down, all cases will be vented to the atmosphere until the ambient case pressure has been reduced to the second test pressure. NOTE: Case pressure change will occur <u>only</u> upon specific ground command.

k. Repeat steps (a) through (i).

l. Repeat step (j) reducing ambient case pressure to the third test pressure.

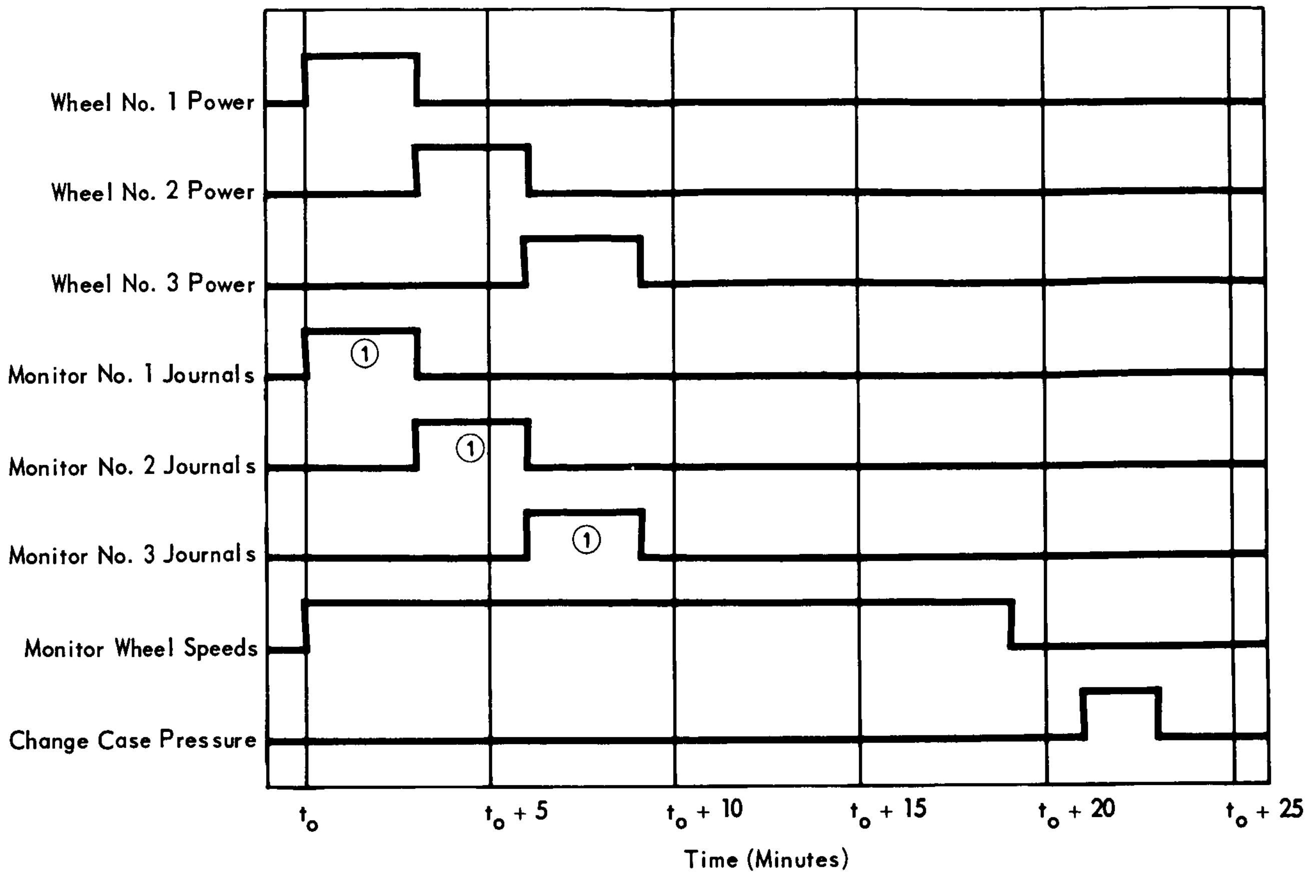
m. Repeat steps (a) through (i).

n. Repeat step (j) reducing ambient case pressure to the fourth test pressure.

o. Repeat steps (d) through (i).

p. Repeat step (j) reducing ambient case pressure to the fifth test pressure.

q. Repeat steps (a) through (i).



NOTES: (1) Automatically monitored with wheel power command.

FIGURE 2.6-5

The Ground Command Test sequence is defined in Figure 2.6-5. This sequence performs the stability test with full voltage applied and run-down test. The stepped-speed stability test is not capable of being performed directly by ground commands because of the complexities introduced into the ground command interface. The flight test will be conducted using the following steps.

a. Power is applied to Wheel No. 1. The No. 1 Journals are monitored automatically in conjunction with power on. All three wheel speeds are monitored sequentially.

b. When Wheel No. 1 reaches full speed, power is removed. Wheel speed monitoring is continued.

c. Repeat steps (a) and (b) on Wheel No. 2.

d. Repeat steps (a) and (b) on Wheel No. 3.

e. When all three wheels have stopped, case pressure is changed to pressure No. 2.

f. Repeat steps (a) through (d).

g. Repeat step (e) arriving at Pressure No. 3.

h. Repeat steps (a) through (d).

i. Repeat step (e) arriving at Pressure No. 4.

j. Repeat steps (a) through (d).

k. Repeat step (e) arriving at pressure No. 5.

l. Repeat steps (a) through (d).

It must be noted that it will not be possible to reverse the case pressure unless a pressure tank and regulator is included with the experimental equipment. It is mandatory that all tests at any fixed pressure level be performed prior to changing pressure.

For each stability test, the following parameters will be time correlated: wheel speed, motor voltage levels, four journal signals, case pressure, and case temperature. The starting and run-down tests for all three wheels will be time correlated. The following parameters will be recorded: three wheel speeds, motor voltage level, twelve journal signals, motor power sequencing signals, case pressures and case temperatures.

The data obtained from this experiment will demonstrate the capabilities of the various air-bearing designs. The results may possibly indicate an advantage of one design over the others. From this plateau, the next logical step will be the development of gas bearing instruments and equipment. Applications range in size from gyroscopes to space power system equipment. In the guidance and control field, interest will be centered on gyroscopes. A logical space testing program for gas bearing gyroscopes should include the following tests.

a. Stability testing of the bearing design.

b. The capability of the gyro to withstand cross-axis rates and accelerations as a function of wheel speed.

c. Drift rates as a function of wheel speed and in the presence of varying degrees of cross-axis loading.

REFERENCES

1. Ausman, J. S., "Linearized ph Stability Theory for Translatory Half-Speed Whirl of Long Self-Acting Gas-Lubricated Journal Bearings", Transactions of the ASME - Journal of Basic Engineering, December 1963

2. Elwell, R. C., Hooker, R. J., and Sternlicht, B., Gas Stability Study - Vertical Rotor Investigation, General Electric Co., General Engineering Laboratory, 20 May 1960, Technical Report for ONR under contract NONr 2844(00).

3. Vohr, J. H. and Chou, C. Y., "Characteristics of Herringbone-Grooved, Gas-Lubricated Journal Bearings", Transactions of the ASME - Journal of Basic Engineering, Paper No. 64-Lub-15.

2.7 Star Characteristics

2.7.1 Objective - The objective of this experiment is to measure the spectral characteristics of the major navigation stars in the pass-bands used by most star trackers and star mappers. The data obtained is intended primarily for application by design engineers as a supplement to existing information on stellar characteristics.

2.7.2 Background - Astronomers have been viewing and observing the stars for thousands of years. During this period of time many techniques have been developed to express the difference in appearance of the individual stars. Presently the difference is expressed in terms of the spectral classification (color) and the magnitude or relative brightness of the stars.

Star magnitude is an arbitrary quantity. If f_1 and f_2 are the fluxes received at the top of the atmosphere from two stars in a given spectral region, their magnitude difference is given by

$$m_2 - m_1 = 2.5 \log (f_1/f_2).$$

The zero point can be chosen quite arbitrarily and the magnitude difference gives relative fluxes. Several kinds of magnitudes are in use - visual (m_v), photographic (m_{pg}), photovisual (m_{pv}), infrared, bolometric and a variety of photoelectric magnitudes. The zero points of the various magnitude scales are adjusted somewhat arbitrarily with respect to each other.

Because of the difference in spectral bands covered by each of the magnitude measuring techniques, comparison of the numbers can provide useful information. As an example, photographic magnitudes are measured in the 0.35μ to $0.5\,\mu$ band while visual magnitudes are obtained in the $0.5\ \mu$ to $0.62\ \mu$ band. If the two measured magnitudes are compared, the relationship will be an indication of the color of the star. The relative color is called the color index factor and is expressed mathematically as:

$$C.I. = m_v - m_{pg}$$

A negative number is an indication of a blue or white hot star and a positive number indicates a cooler or red star.

Spectral classification (S_p) is used to divide the stars into several classes according to temperature and color. The more common classifications are B, A, F, G, K and M. Each class is further subdivided into ten subclasses from 0 to 9 to express variations in each class. In addition, the stars are divided into groups according to their intrinsic or absolute magnitudes (Reference 1). The main groups most commonly used are

- I Super Giants (including Class C stars)
- II Bright Giants
- III Giants
- IV Subgiants
- V Main Sequence (also called dwarfs)
- S_d Subdwarfs
- W White Dwarfs

To indicate the usage of the classes and groups of stars and their relationship, Table 2.7-1 contains a listing of the major groups and some of the classes with the color indices, effective temperature, wavelength of maximum radiation and relative distribution. Values for other groups and classes may be approximated by extrapolation from the values given in the table. The table illustrates the previously mentioned negative color index for the hot stars and also indicates that as the temperature goes up the wavelength at which radiation is maximum goes down. The relationship of maximum radiation wavelength to temperature is known as Wein's displacement law and is expressed as

$$\lambda_m = \frac{2893}{T} \quad \text{(in microns)}$$

TABLE 2.7-1
SPECTRAL AND LUMINOSITY CLASSES

Sp	MAIN SEQUENCE, V			GIANTS, III			SUPERGIANTS, I			DISTRIBUTION
	C.I.	T_e	λ_m	C.I.	T_e	λ_m	C.I.	T_e	λ_m	
		°K	μ		°K	μ		°K	μ	%
B0	−0.42	22,000	0.13							32
B5	−0.36	14,000	0.206							
A0	−0.17	10,700	0.27							33
A5	+0.03	8,500	0.34							
F0	+0.15	7,400	0.39	+0.22						9
F5	+0.28	6,500	0.445	+0.40			+0.34			
G0	+0.42	5,900	0.49	+0.60	5200	0.555	+0.70	4900	0.59	7
G5	+0.58	5,500	0.525	+0.82	4600	0.630	+1.05	4300	0.675	
K0	+0.77	4,900	0.59	+1.06	4100	0.705	+1.40	3800	0.762	17
K5	+1.08	4,200	0.69	+1.35	3600	0.805		3300	0.877	
M0	+1.30	3,600	0.805	+1.50	3400	0.850		3000	0.965	7
M5	+1.50	2,800	1.03		2800	1.03				

Effective temperature, color indices, wavelength for maximum radiation and distribution of the 100 brightest stars ($m_v < +2.5$) by class.

Figure 2.7-1 clearly shows the relationship of radiance and wavelength as a function of temperature.

The temperatures given are the equivalent black body temperatures. Although a star may be a main sequence hot star, the flux received at or near the earth may be relatively low. The power or flux received from a remote point source on a small broad band collector is proportional to the absolute temperature to the fourth power and inversely proportional to the distance to the source squared.

$$P = k\ (T^4/d^2)$$

Navigation star temperatures range from 3000°K to 27,000°K and are at a distance ranging from about 4 light years to 2600 light years from the earth. One light year is 5.9×10^{12} miles.

One of the more extensively used photoelectric identification techniques is one developed by Johnson and Morgan (Reference 2). Three separate filters are placed over a 1P21 photomultiplier detector and the magnitudes of the star in the ultraviolet, blue and visible ranges are measured. The magnitudes are commonly

BLACK-BODY RADIATION CURVES FOR VARIOUS TEMPERATURES

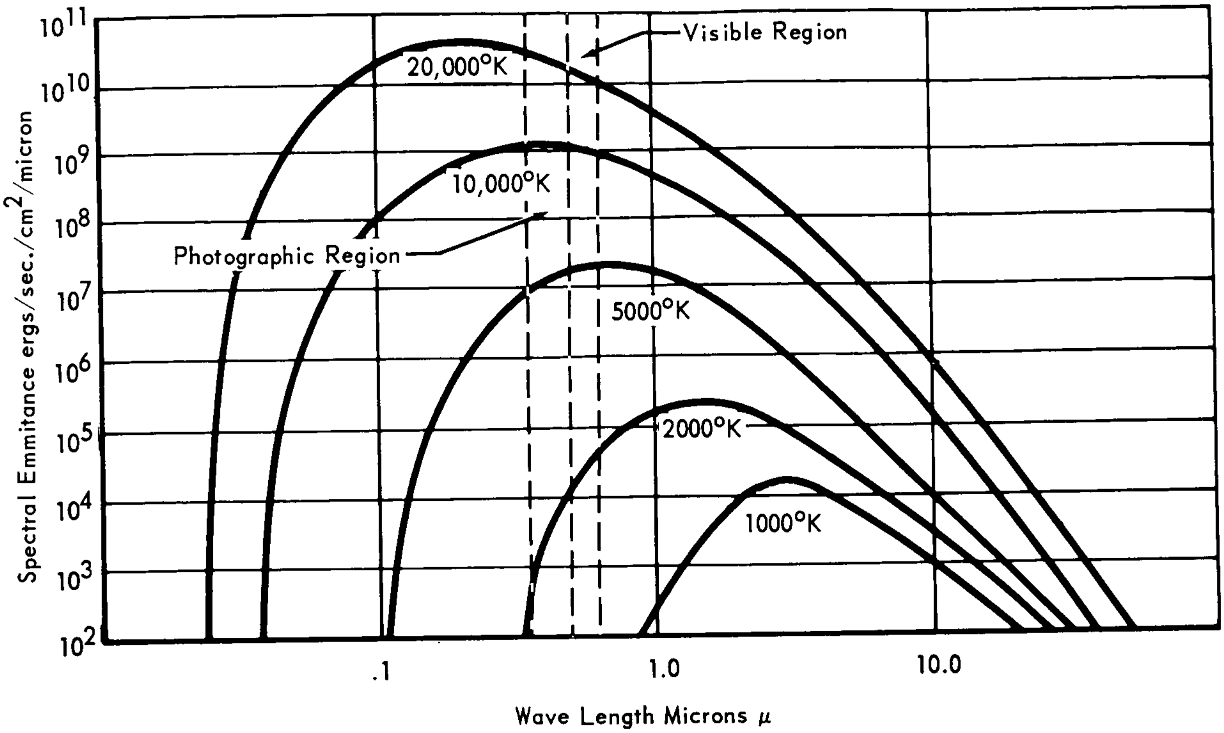


FIGURE 2.7-1

known as the UBV color magnitudes. Figure 2.7-2 illustrates the relative pass bands of the three filters when used in conjunction with a 1P21 photomultiplier with an S-4 photo-emissive surface.

Utilizing the Johnson-Morgan system provides magnitude and color recognition information. The ratio of the B and V magnitudes is a direct function of the star color. Star lists for tracker designers commonly list the V (visible) magnitude and the B-V magnitude to indicate color. Use of a color recognition technique has been proposed for star recognition in star tracker systems (Reference 3).

Star tables normally list the star magnitudes as seen from the top of the atmosphere. This requires that corrections for atmospheric attenuation be applied to surface made measurements. Normally the correction factor used is

$$m = m_o = a_\lambda \text{ (sec Z)}$$

where m = magnitude of light received on earth's surface

m_o = magnitude of light as viewed outside

a_λ = extinction coefficient

Z = angle of viewing the light source with respect to local vertical.

Typical value for the extinction coefficient used for the Johnson-Morgan technique measurements are 0.59 for the U filter, 0.25 for the B filter and 0.10 for the V filter. These values are approximate and a function of the atmospheric conditions. Because of the day to day variation in the atmospheric conditions for viewing, a variation in the measured star magnitude exists. The tables list a statistical average and a variation of 5% to 10% between tables is not uncommon. This amount of variation may be insignificant to the human observer who can easily use star pattern

SPECTRAL RESPONSE OF IP21 PHOTOMULTIPLIER WITH U – B – V FILTERS

(JOHNSON AND MORGAN TECHNIQUE)

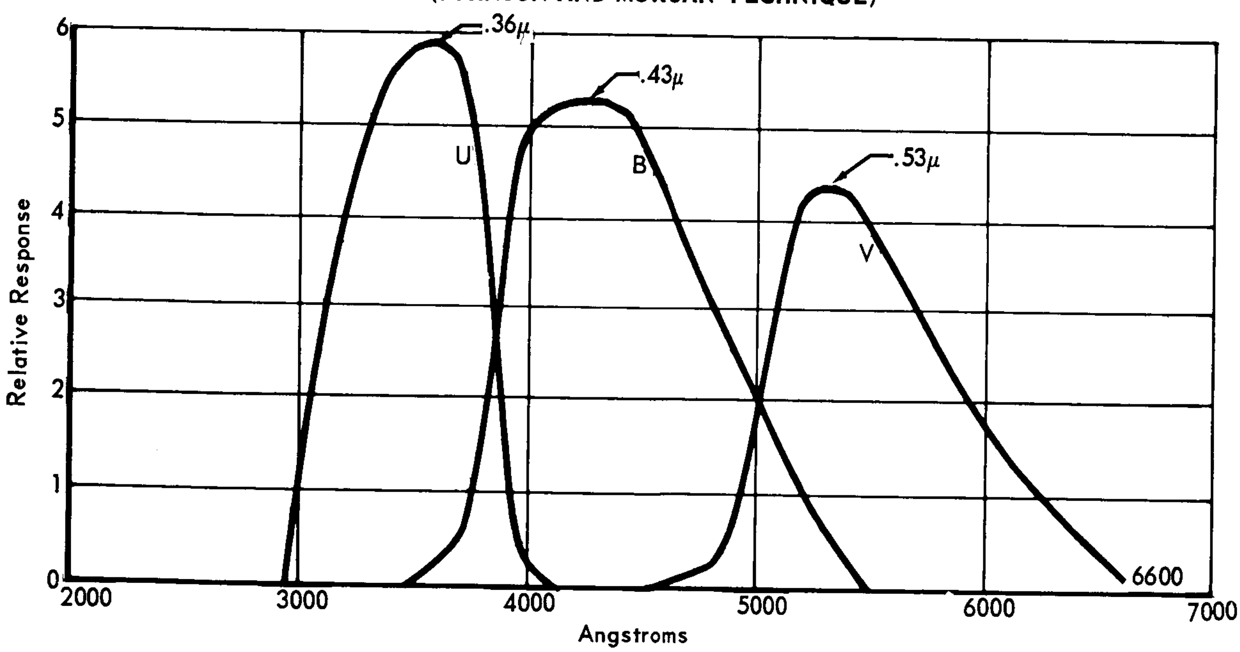


FIGURE 2.7–2

and constellations to recognize particular stars, but to the designer of star trackers or star mappers and the associated computers, the problem of recognition and magnitude variation is significant.

Figure 2.7-3 is a plot of the approximate number of stars brighter than a given magnitude as a function of visual magnitude. It is apparent that variations in magnitude, changes in resolution and variation in sensitivity of a detector will affect the number of stars detected by a given sensor design. Referring to the

STAR MAGNITUDE DISTRIBUTION

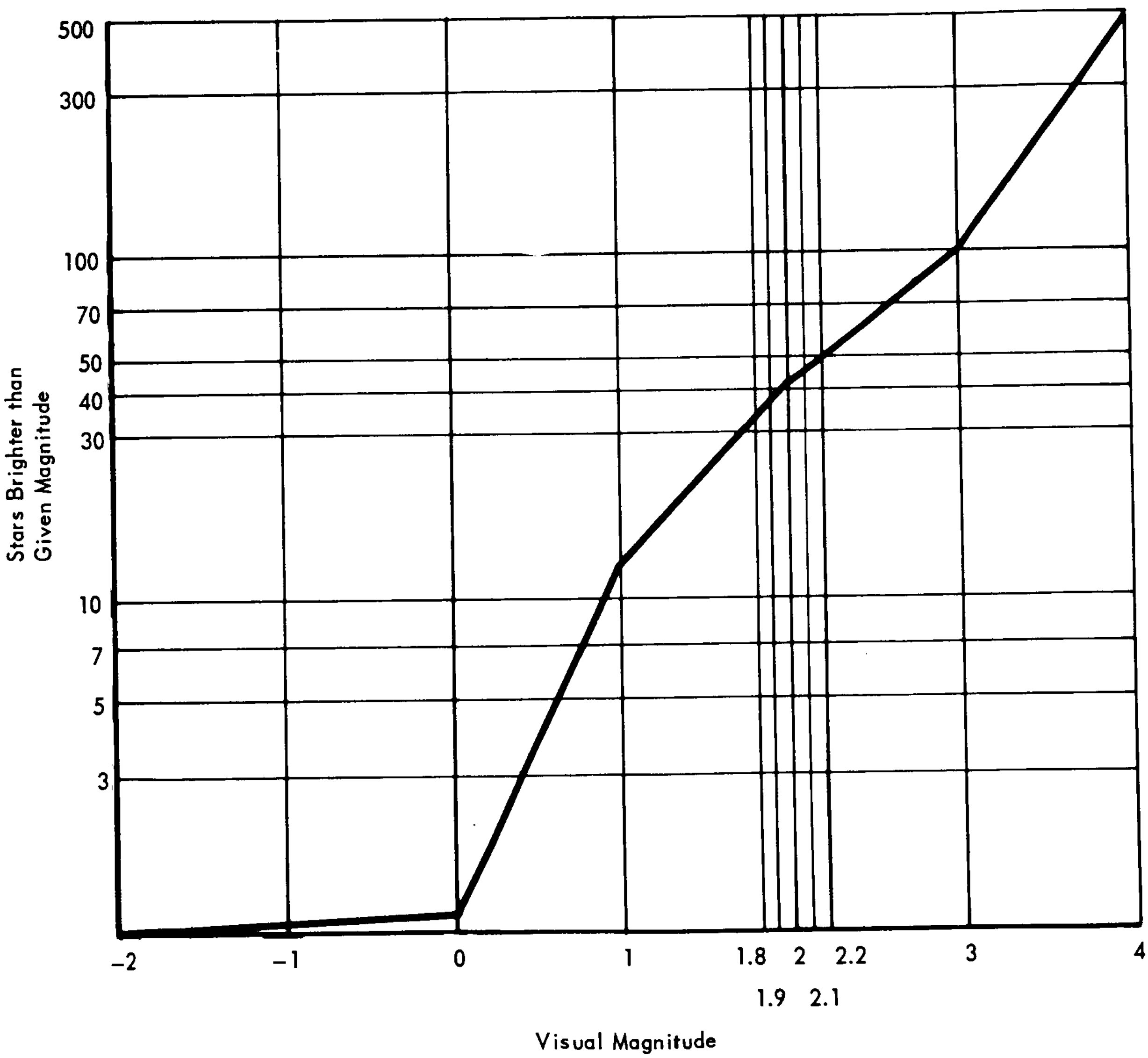


FIGURE 2.7–3

plot, a variation of 10% in apparent magnitude about a level of 2.0 gives a change in the number of detectable stars from 30 to 50 with a nominal value of 40 stars. In addition, a variation in detector sensitivity will increase or decrease the number of detectable stars. As an example, assume a tracker is designed and adjusted to a threshold of +2.8 magnitude, i.e., marginal tracking with a minimal signal to noise ratio can occur if the star is brighter than +2.8. The possibility of an 0.2 variation in star magnitude indicates that stars of +2.9, which were assumed below the detection level, may actually be +2.7 and therefore trackable. The additional trackable stars can result in incorrect output indications of vehicle position coordinates unless the computer storage capacity is increased.

Based on the calculated star magnitude as seen outside the atmosphere and with a knowledge of the spectral characteristics of the star, the sensitivity of photomultiplier or photodetector tubes may be calculated. The relative response is indicated by values of magnitudes known as Color Corrected Magnitudes (CCM). The CCM will vary with detector types and with the photoemissive surface used. Figure 2.7-4 illustrates the relative response of four types of detectors as a function of wavelength. The variation in spectral pass band, peak response and sensitivity is clearly illustrated. The curves shown are nominal and each specific detector will show some deviation from the curves given. Each type of detector will present a different response to each of the stars, depending upon the star's magnitude and spectral distribution. Stars clearly visible to a vidicon detector may not produce an output from a photo detector with a S-11 surface.

Table 2.7-2 is a listing of the 100 brightest (visible) stars. The stars are listed in accordance with their right ascension (α) and declination (δ) positions, Epoch 1900. The data provided for each star is the position, visible magnitude (V), ratio of blue and visible magnitude (B-V) and the spectral classification. In

PHOTODETECTOR RESPONSE

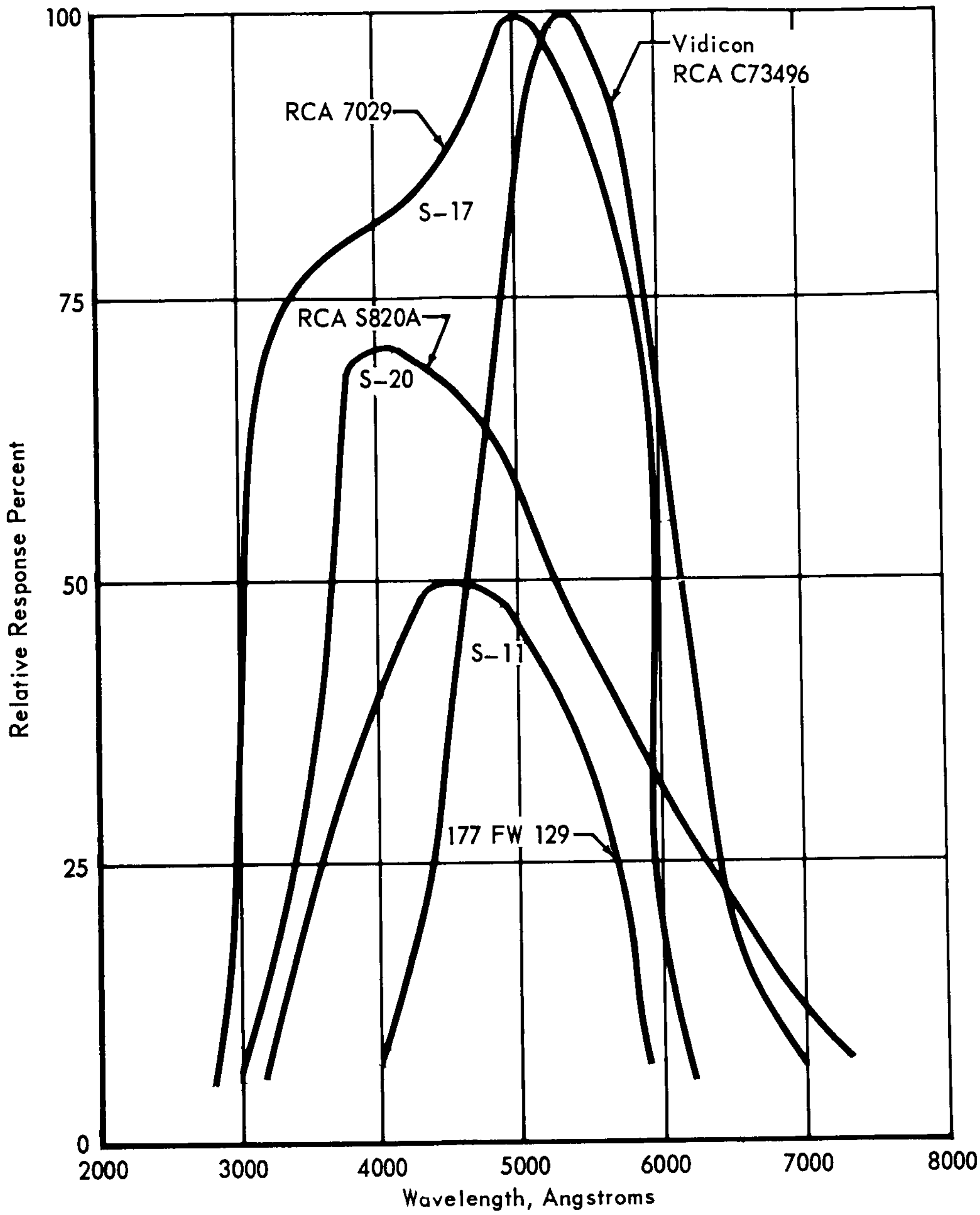


FIGURE 2.7-4

TABLE 2.7–2
THE BRIGHTEST STARS

STAR		1900				MAGNITUDE				Sp
		α		δ		V	B−V	C.C.M.		
		h	m	°	′			S−11	S−20	
Alpheratz	α And	0	03	+28	32	2.07	−0.07	2.22	2.15	B9p III
Caph	β Cas	0	04	+58	36	2.26	+0.34	–	–	F2 IV
Ankaa	α Phe	0	21	−42	51	2.37	+1.07	–	–	K0 III
Schedar	α Cas	0	35	+55	59	2.20	+1.16	–	–	K0 II–III
Diphda	β Cet	0	39	−18	32	2.04	+1.01	2.86	–	K0 III
Cih	γ Cas	0	51	+60	11	2.15	−0.2	–	–	B0e IV
Mirach	β And	1	04	+35	05	2.07	+1.62	–	–	M0 III
Polaris	α UMi	1	23	+88	46	2.02	+0.6	–	–	F8 Ib
Achernar	α Eri	1	34	−57	45	0.49	−0.17	0.39	0.42	B5 IV
Almach	γ And	1	58	+41	51	2.16	+1.3	–	–	K3 II–III
Hamal	α Ari	2	02	+22	59	2.00	+1.17	–	–	K2 III
Mira	ο Cet	2	14	− 3	26	2.0	+1.5	–	–	M6e III
Menkar	α Cet	2	57	+ 3	42	2.53	+1.16	–	–	M2 III
Algol	β Per	3	02	+40	34	2.10	−0.05	2.09	1.99	B8 V
Mirfak	α Per	3	17	+49	30	1.80	+0.48	–	–	F5 Ib
Aldebaran	α Tau	4	30	+16	19	0.80	+1.55	2.04	1.95	K5 III
Capella (9)	α Aur	5	09	+45	54	0.09	+0.81	0.76	0.73	G8, G0
Rigel	β Ori	5	10	− 8	19	0.11	−0.05	0.05	0.23	B8 Ia
Bellatrix	γ Ori	5	20	+ 6	18	1.63	−0.22	1.44	1.43	B2 III
El Nath	β Tau	5	20	+28	32	1.65	−0.13	1.54	1.67	B7 III
Mintaka	δ Ori	5	27	− 0	22	2.19	−0.21	1.98	2.21	O9.5 II
Arneb	α Lep	5	28	−17	54	2.58	+0.22	–	–	F0 Ib
Alnilam	ε Ori	5	31	− 1	16	1.70	−0.18	1.54	1.42	B0 Ia
Alnitak	ζ Ori	5	36	− 2	00	1.79	−0.21	1.59	1.72	O9.5 Ib
Saiph	κ Ori	5	43	− 9	42	2.06	−0.16	1.88	1.87	B0.5 Ia
Betelgeuse	α Ori	5	50	+ 7	23	0.4	+1.85	–	–	M2 I
Menkalinan	β Aur	5	52	+44	56	1.89	+0.04	1.93	2.30	A2 V
Mirzam	β CMa	6	18	−17	54	1.96	−0.23	1.77	1.69	B1 II–III
Canopus	α Car	6	22	−52	38	−0.72	+0.16	−0.6 ′	−0.55	F0 I–II
Alhena	γ Gem	6	32	+16	29	1.93	0.00	1.93	1.93	A0 IV
Sirius	α CMa	6	41	−16	35	−1.44	−0.01	−1.46	−1.58	A1 V
Adhara	ε CMa	6	55	−28	50	1.48	−0.17	1.34	1.33	B2 II
Wezen	δ CMa	7	04	−26	14	1.85	+0.63	2.40	–	F8 Ia
Aludra	η CMa	7	20	−29	06	2.42	−0.07	2.34	2.25	B5 Ia
Castor	α Gem	7	28	+32	06	1.56	+0.05	1.63	1.58	A1, A5
Procyon	α CMi	7	34	+ 5	29	0.36	+0.41	0.72	0.90	F5 IV–V
Pollux	β Gem	7	39	+28	16	1.15	+1.01	2.00	1.99	K0 III
Naos	ζ Pup	8	00	−39	43	2.23	−0.27	1.97	1.74	O5
	γ Vel	8	06	−47	03	1.85	−0.25	1.59	1.39	WC7
Avior	ε Car	8	20	−59	11	1.94	+1.2	–	–	K0, B
	δ Vel	8	42	−54	21	1.93	+0.04	1.95	2.01	A0 V
Suhail	λ Vel	9	04	−43	02	2.23	+1.7	–	–	K5 Ib
Miaplacidus	β Car	9	12	−69	18	1.68	−0.01	–	–	A0 III
Scutulum	ι Car	9	14	−58	51	2.24	+0.18	–	–	F0 Ib
	κ Vel	9	19	−54	35	2.45	−0.16	2.30	–	B2 IV
Alphard	α Hya	9	23	− 8	14	2.05	+1.43	–	–	K4 III
Regulus	α Leo	10	03	+12	27	1.34	−0.11	1.26	1.23	B7 V
Algeiba	γ Leo	10	14	+20	21	2.02	+1.2	–	–	K0p III
Merak	β UMa	10	56	+56	55	2.36	−0.02	–	–	A1 V
Dubhe	α UMa	10	58	+62	17	1.81	+1.06	–	–	G9 III
Zosma	δ Leo	11	9	+21	04	2.55	+0.12	2.66	–	A4 V
Denebola	β Leo	11	44	+15	08	2.13	+0.08	2.22	2.28	A3 V
Phecda	γ UMa	11	49	+54	15	2.43	0.00	–	–	A0 V
Gienah	γ Crv	12	11	−16	59	2.58	−0.09	2.50	–	B8 III
Acrux	α Cru	12	21	−62	33	0.83	−0.26	0.66	1.28	B1, B3
Gacrux	γ Cru	12	26	−56	33	1.68	+1.58	–	–	M3 II
Muhlifain	γ Cen	12	36	−48	25	2.16	−0.01	2.13	–	A0 IV
Mimosa	β Cru	12	42	−59	09	1.29	−0.25	1.11	1.20	B0 III
Alioth	ε UMa	12	50	+56	30	1.78	−0.02	–	–	A0p
Mizar	ζ UMa	13	20	+55	27	2.12	+0.03	–	–	A2 V
Spica	α Vir	13	20	−10	38	0.97	−0.23	0.80	0.94	B1 V
	ε Cen	13	34	−52	57	2.34	−0.23	2.08	2.26	B1 IV
Alcaid	η UMa	13	44	+49	49	1.86	−0.19	–	–	B3 V
Hadar	β Cen	13	57	−59	53	0.63	−0.24	0.47	0.56	B1 II
Menkent	θ Cen	14	01	−35	53	2.07	+1.02	2.80	–	K0 III–IV
Arcturus	α Boo	14	11	+19	42	−0.05	+1.24	0.94	1.02	K1 III
	η Cen	14	29	−41	43	2.39	−0.21	2.16	–	B2 V
Rigil Kent	α Cen	14	33	−60	25	−0.27	+0.71	0.29	0.85	G2, K1
	α Lup	14	35	−46	58	2.5	−0.22	2.09	–	B1 V
Izar	ε Boo	14	41	+27	30	2.39	+0.93	–	–	K1, A
Kochab	β UMi	14	51	+74	34	2.04	+1.49	–	–	K4 III
Alphecca	α CrB	15	30	+27	03	2.22	−0.02	2.21	2.31	A0 III
Dzuba	δ Sco	15	54	−22	20	2.32	−0.14	2.18	2.21	B0 V
Acrab	β Sco	16	00	−19	32	2.52	−0.09	–	–	B0.5 V
Antares	α Sco	16	23	−26	13	0.94	+1.83	2.33	–	M1, B
	ζ Oph	16	32	−10	22	2.56	0.00	2.58	–	O9.5 V
Atria	α TrA	16	38	−68	51	1.93	+1.43	–	–	K2 III
	ε Sco	16	44	−34	07	2.29	+1.15	–	–	K2 III–IV
Sabik	η Oph	17	05	−15	36	2.44	+0.05	2.50	–	A2.5 V
Shaula	λ Sco	17	27	−37	02	1.60	−0.23	1.42	1.44	B1 V
	θ Sco	17	30	−42	56	1.86	+0.38	2.19	2.35	F0 Ib
Ras-Alhague	α Oph	17	30	+12	38	2.07	+0.15	2.21	2.33	A5 III
	κ Sco	17	36	−38	59	2.39	−0.21	2.17	2.44	B2 IV
Eltanin	γ Dra	17	54	+51	30	2.21	+1.54	–	–	K5 III
Kaus Australis	ε Sgr	18	18	−34	26	1.81	−0.02	1.78	1.95	B9 IV
Vega	α Lyr	18	34	+38	41	0.03	0.00	0.04	0.14	A0 V
Nunki	σ Sgr	18	49	−26	25	2.09	−0.20	1.90	1.92	B2 V
	ζ Sgr	18	56	−30	01	2.57	+0.09	2.66	–	A2 IV
Altair	α Aql	19	46	+ 8	36	0.77	+0.22	0.99	1.08	A7 IV–V
Peacock	α Pav	20	17	−57	03	1.94	−0.20	1.78	1.90	B3 IV
Sadir	γ Cyg	20	19	+39	56	2.22	+0.66	2.78	–	F8 Ib
Deneb	α Cyg	20	38	+44	55	1.25	+0.08	1.34	1.38	A2 Ia
Gienah	ε Cyg	20	42	+33	36	2.46	+1.03	–	–	K0 III
Alderamin	α Cep	21	16	+62	10	2.43	+0.23	–	–	A7 IV–V
Enif	ε Peg	21	39	+ 9	25	2.38	+1.56	–	–	K2 Ib
Al Na'ir	α Gru	22	02	−47	27	1.75	−0.14	1.63	1.98	B5 V
	β Gru	22	37	−47	24	2.16	+1.62	–	–	M3 III
Fomalhaut	α PsA	22	52	−30	09	1.16	+0.09	1.24	1.40	A3 V
Scheat	β Peg	22	59	+27	32	2.50	+1.7	–	–	M2 II–III
Markab	α Peg	23	00	+14	40	2.49	−0.04	2.45	–	B9.5 III

LEGEND:

- V – Visible magnitude, Morgan-Johnson UBV Scale
- B−V – Ratio of Blue and Visible magnitudes – Indicates color
- Sp – Spectral Classification of Star. See Table 27–1 for further information on symbols
- C.C.M. – Color corrected magnitudes, response of photo-emmisive detectors with S–11 and S–20 photosensitive surfaces

addition, the color corrected magnitude (CCM) for two types of detectors is given on the major navigation stars.

The wide range of variables involved in star magnitude and spectrum and the detector variations of threshold, sensitivity and gain complicate not only the initial design of a star tracker but also the ground test evaluation. Star magnitude simulation must be over a range equivalent to the known uncertainty in the star magnitudes. Some attempt must be made to reproduce proper spectral ranges for the simulated stars; however, exact color simulation is extremely difficult. To minimize the complexity of ground simulation, a reduction in the number of variables is needed. This requires a more detailed knowledge of the navigation star magnitudes, colors and background noise in the detector spectral band. A measurement from orbit of the magnitude and color of several selected navigation stars in the 3000 to 7000 angstrom spectral band is highly desirable. The Orbiting Astronomical Observatory (OAO) is designed to perform a detailed mapping of the celestial sphere in the 1200 to 2900 angstrom band and does not, at present, include the spectral range most commonly used by star mappers/star trackers.

Several techniques could be used to obtain the desired information. A general discussion of some of these techniques is contained in the following paragraphs.

One method which could be used would be to add a star tracker to a vehicle with the capability of being pointed to selected stars. After acquisition and tracking have commenced, the detector current, detector applied voltage and the AGC (Automatic Gain Control) voltage would be measured. Through use of calibration curves and data reduction techniques the magnitude of the star could be calculated. This method will not provide a color measurement. The method is not very accurate for magnitude determination due to the non-linearities involved in detector gain as a function of applied voltage which is varied or adjusted by the AGC controlling the detector current range for tracking purposes.

The second method would be to utilize a broad band or narrow band grating spectrometer to obtain the data. This technique will provide very good data, but is very complex. Usage requires precision pointing of the device. Gimballing of the device for pointing is complex and large due to the size of most of the available devices. One representative detector is 14 inches in diameter, 30 inches long and weighs 100 pounds for a rigidly mounted unit. Gimballing could more than double the size.

A third method which could be considered would be to mount a telescope/detector assembly rigidly to the body of a vehicle with a star tracker master reference. The detector would be a 1P21 photodetector with a three-color wheel for Johnson-Morgan measurements. The vehicle would point the detector to the star being tracked by aligning the detector along the tracker line of sight at tracker gimbal null position. After the tracker acquires a star and starts tracking, the vehicle is slewed to null the gimbal outputs. When the gimbals are at null, the detector is energized and a measurement is made of the tracked star.

The fourth, and recommended, method is to replace the detector assembly in a OAO star tracker with a three-color detection system. The modified tracker would be slaved to the master reference star tracker. The experiment detector will then always point to the same area as the master reference within the limits of slaving accuracy. Each time the master reference star tracker acquires and tracks a star, the slaved unit measures the characteristics of the tracked star and area around the star. This technique will not increase the complexity of vehicle attitude requirements since vehicle slewing is not required.

2.7.3 Functional Description - Figure 2.7-5 is a functional block diagram of the experimental package for measuring the characteristics of the major navigation stars. A photodetector assembly is mounted on the inner gimbal of a modified OAO

FUNCTIONAL BLOCK DIAGRAM
STAR CHARACTERISTICS EXPERIMENT

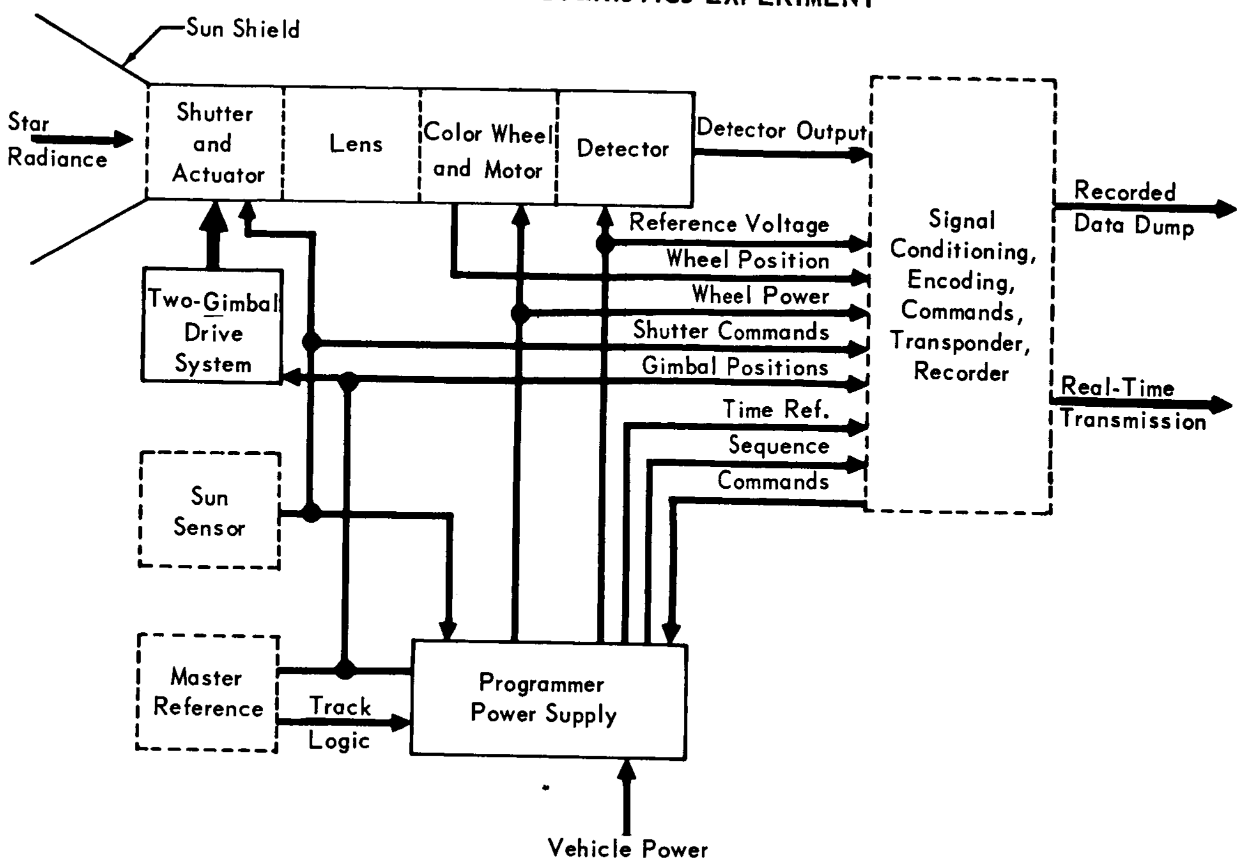


FIGURE 2.7-5

star tracker. The gimbals are slaved to the gimbal positions of the master reference star tracker. The characteristics of the star are determined through use of the three-color system devised by Johnson and Morgan.

Detector Assembly - A photomultiplier tube and the associated optics and filters are mounted on the inner gimbal of a modified star tracker. The optics consist of a 1° field-of-view lens system, a sun shutter and a three-color filter wheel. The three-color filter plus detector have spectral characteristics as shown in Figure 2.7-2. The system is operated open-loop, i.e., the detector is not involved in a tracking loop and has a constant voltage applied.

When the sun shutter is open and the master reference is tracking a star, the output of the detector will be a measure of the magnitude of the star being tracked as well as the effect of stellar background "noise". When the shutter is closed (primarily to protect the photodetector from damage due to solar radiation) the current of the detector will be a measure of the relative condition and gain of the system (dark current).

The three-color Johnson-Morgan technique is used because (a) direct correlation of data with a large catalog of ground based measurements, (b) simplicity of design, and (c) ease of instrumenting for space-to-ground data transmission. The data obtained can be readily converted to Color Corrected Magnitudes (CCM) for several different types of photodetectors.

Test Method - The detector unit is mounted in a slaved gimbal system to provide the required pointing to the guide or navigation stars. After vehicle orbit insertion and alignment the master reference star tracker is pointed to a guide star. During the track period of the master reference, the color wheel of the experiment package is rotated and the output from the detector is monitored. The photomultiplier output current, applied voltage and wheel position are recorded. In addition, the star tracker gimbal positions, time reference and other selected parameters are recorded. Even though the star tracker may track one star for up to 20 minutes, only 1 to 2 minutes of data is needed on each star. This keeps the recording from becoming a major problem inasmuch as only 8 to 10 minutes of data is recorded per orbit.

Measurements of about 10 stars are taken and the data is correlated with presently available ground based measurements. If the correlation is good, say within 2% to 5%, the experiment is turned off. If the correlation is not good, the experiment is continued until 50 or more navigation stars are mapped.

Major Error Sources - The major error sources in determining the magnitude of the navigation stars are changes in detector sensitivity, degradation of optics, nonlinearities and drift of the instrumentations and the effects of misalignment of the test unit. Misalignment introduces errors caused by the variation of background stellar noise with differences in field-of-view. Use of a 1° field-of-view for the detector allows up to 0.1° error in the system which is within the state-of-the-art limitations for slaving and alignment. A "retest" of several of the navigation stars after a period of time will provide a measure of the system variation.

2.7.4 Experiment Physical Parameters - Table 2.7-3 contains a tabulation of the experiment package physical parameters. The parameters of the OAO star tracker are included in order than an estimate of the experiment total size is available.

2.7.5 Data Parameters - Table 2.7-4 is a tabulation of the experiment parameters to be telemetered for this experiment. The relative need for the parameter

TABLE 2.7-3

STAR CHARACTERISTICS EXPERIMENT PHYSICAL PARAMETERS

ITEM	VOLUME		WEIGHT (LB.)	AVG. POWER (WATTS)	
	CU.FT.	L – W – H (INCHES)		PEAK	NOMINAL
Experiment Sensor Sensor Head Electronics	 .88 .14	 17 x 11 x 8 15 x 4 x 4	 23.5 7.5	10	9
Support Equipment Programmer Signal Conditioner	 .02 .01	 4 x 3 x 3 2 x 3 x 3	 1 0.5	 2 .5	 1 .5
Subtotal	1.05		32.5	12.5	10.5
Master Reference OAO Star Tracker	 3.3	 18 x 17 x 19	 45	 20	 15.4
Total	4.35		77.5	32.5	25.9

TABLE 2.7–4
STAR CHARACTERISTICS EXPERIMENT DATA PARAMETERS

DATA POINT	SIGNAL FORMAT	RANGE OF PARAMETER	SAMPLE RATE (FREQ. RESP.)	ACCURACY	RECORD	ESSEN-TIAL	DESIR-ABLE	CON-TINUOUS
Photomultiplier								X
Current	Analog	0–5 V	1 KC	± 1%	X	X		X
Applied Voltage	Analog	1000 V	DC	± 1%	X	X		X
Filter Position	Digital	pulse	1 pulse/sec		X	X		
Shutter	Digital	on/off			X	X		X
Temperature	Analog	0–160°F		5%	X		X	
*Master Reference								
Gimbal No. 1	Digital		10 cps	2%	X	X		X
Gimbal No. 2	Digital		10 cps	2%	X	X		X
Input Power	Analog		DC	5%			X	
Temperature	Analog	0–160°F		5%			X	
*Secondary Reference								
Vehicle Pitch	Analog		10 cps	2%	X	X		
Vehicle Roll	Analog		10 cps	2%	X	X		
Vehicle Yaw	Analog		10 cps	2%	X	X		
Miscellaneous								
*Time Reference	Digital	22 bits		± 1 bit	X	X		
*Vehicle Rates (3)	Analog				X		X	
*Spacecraft Power Buss	Analog				X		X	

Spacecraft Parameters are marked (*)

is indicated in the desired or essential columns. Additional parameters may be added for real time transmission to aid in trouble and failure analysis.

2.7.6 <u>Vehicle Orbit and Attitude Requirements</u> - In order to obtain maximum useful data, certain vehicle parameters must be controlled. The orbit parameters are not critical; however, an equatorial orbit provides the greatest star coverage (the navigation stars tend to be clustered primarily about the galactic equator). Small deadbands and low rates are essential for this experiment. Rates in excess of 0.1°/sec and a deadband greater than $\pm$ 2° in attitude control increase the control requirements on the star tracker excessively. A vehicle with three-axis control, either inertially or earth-oriented may be used as the carrier for this experiment. An earth-oriented vehicle with a sun sensor provides a larger star field coverage and easier data reduction than an inertially oriented vehicle.

2.7.7 <u>Equipment Support and Data Handling Requirements</u> - This experiment is designed to involve as little additional load on the spacecraft equipment as possible. The programmer requirements are minimal and utilize as much as possible the master reference system signals. The general requirements for a master attitude reference system are discussed in Appendix A. The general requirements for Data Handling are discussed in Appendix B.

<u>Power Conversion and Filters</u> - In order to provide a calibrated reference voltage for the photodetector, a very stable high voltage converter is required. Stabilization to better than 1% is required. This will require a well filtered, stabilized input voltage from the spacecraft or a stabilizing network in the detector power source assembly.

<u>Programmer</u> - This unit contains the necessary electronics, relays, timers and actuators to provide the following functions:

a. Turn the experiment on and off at predetermined times or upon ground command.

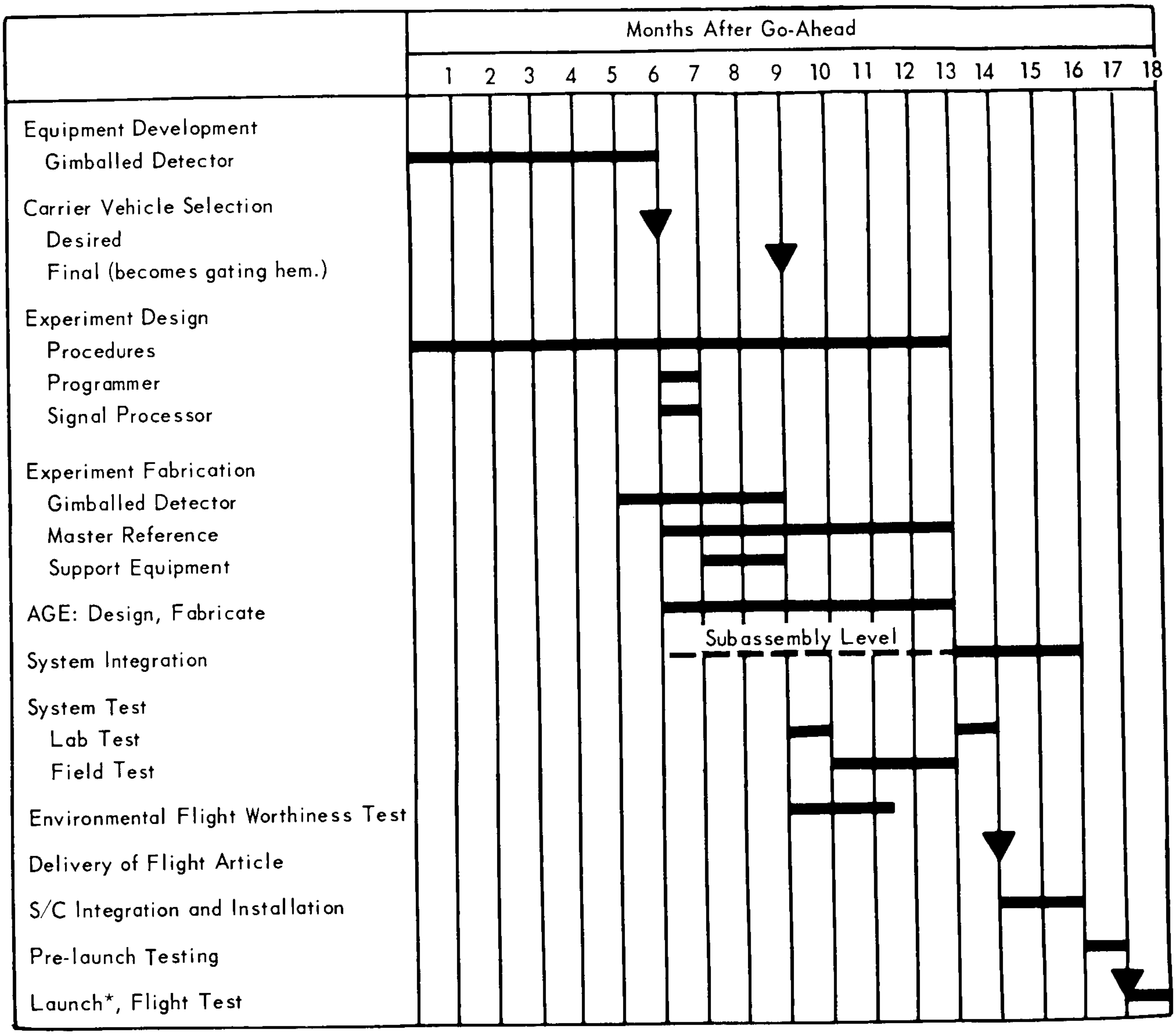
b. Actuate the sun shutter at predetermined intervals, upon ground command and upon receipt of a sun sensor output indicating that the sun is within 30° of the detector field-of-view.

c. Provide the logic to turn the recorder on during the master reference track periods when the experiment is on.

d. Provide a time reference.

<u>Signal Conditioning</u> - Except for the requirement for high precision data transmission, signal conditioning should not present any special problems. It may be desirable to use either a non-linear amplifier or three separate channels to provide a high gain at low detector currents, medium gain at medium currents and low gain at high currents.

STAR CHARACTERISTIC EXPERIMENT DEVELOPMENT PLAN



*Depends on carrier vehicle – may vary from 1 month to 8 months after installation.

FIGURE 2.7–6

Environmental Control - The environmental requirements of this experimental package are essentially the same as the OAO star tracker. Use of the 1P21 photo-detector in the experiment relieves the temperature limitations imposed by some other types of detectors.

Electrical Energy Requirements - If it is assumed that 50 stars are to be measured and that approximately half power is required between measurements, a power calculation can be made. Using the 12.5 watt figure for two minutes on 50 stars gives 20 watt-hours. Using 6 watts (standby-power) between measurements and assuming operation in standby for 17 hours gives 102 watt-hours. This can be reduced considerably by reducing the standby power or turning the experiment off between stars.

2.7.8 Experiment Flight Test Plan - The overall experiment development plan is shown in the flow diagram of Figure 2.7-6. The schedule is based on an estimate of the hardware development, ground tests, system integration and interfacing, calibration and flight time. The schedule is intended for planning purposes.

Of special note in the schedule is the six months allowed for development of the gimballed detector system and the three months allowed for field tests. The field tests consist of measuring the star characteristics from a ground test site using either the flight unit or one identical to the flight unit.

Orbital Test Sequence - The operational test sequence is relatively simple. After the star lock-on is accomplished by the master reference, the experiment is turned on a 1 to 2 minute recording of the star data is made. The sequence is repeated each time the star tracker acquires and tracks a star until the 50 major navigation stars are measured and analyzed.

REFERENCES

1. C. W. Allen, "Astrophysical Quantities", Athlone Press, London, England, 1957

2. H. L. Johnson and W. W. Morgan, "Fundamental Stellar Photometry," Astrophysical Journal, Vol: 117, p 313, 1953

3. J. H. Flink, "Star Identification by Optical Radiation Analysis", IEEE Transactions on Aerospace and Navigation Electronics, Vol. ANE-10, No. 3, pp 212 to 221, September 1963.

3. CATEGORY B EXPERIMENTS

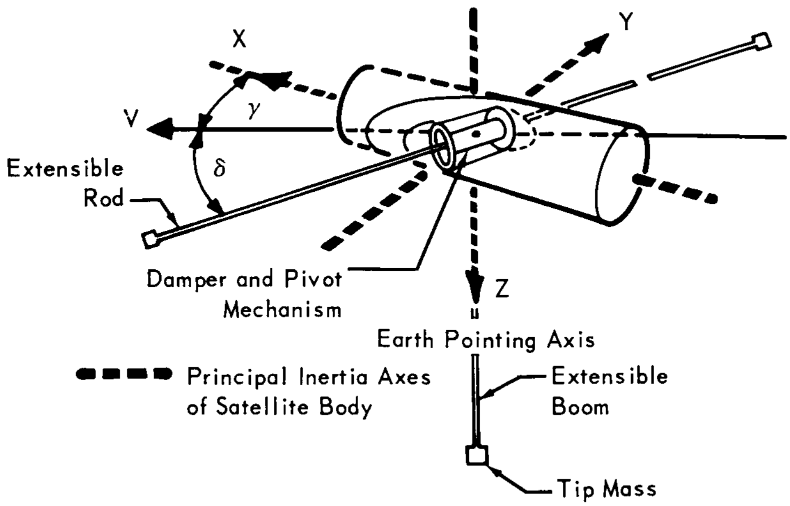
3.1 Gravity Gradient Controls - Passive Damping

3.1.1 Objective - The purpose of this experiment is to use a satellite configured with an extensible boom and damper for passive stabilization by inertia coupling to: (1) determined the accuracy with which a satellite in a low altitude earth orbit may be aligned in three axes by an external torque due to the gradient of the gravitational field, (2) obtain design data for this and other approaches. If demonstrated to achieve the desired alignment accuracy of 1 to 3 degrees gravity gradient stabilization with passive damping has great potential application for long life near earth orbiting vehicles.

3.1.2 Background - The earth's gravitational field provides torques which tend to align the axis of least moment of inertia of a rigid body along the local vertical, the axis of second least moment of inertia in the orbit plane, and the axis of highest moment of inertia perpendicular to the orbit plane. For applications in which a satellite local vertical orientation is required, the principal moment of inertia about the earth pointing axis must be considerably less than about any transverse axis. Since such a favorable relationship between the inertia axes rarely exists in present day satellites, an artificial means of producing the desired inertia ratios is employed. A boom may be deployed along the desired earth pointing axis to create the required inertia configuration, and a secondary rod coupled to the satellite by a damper mechanism extended for use as an auxiliary body to obtain passive damping. Boom and damper physical parameters are dependent upon satellite design and mission requirements.

Two axis gravity gradient stabilization of an earth satellite has been demonstrated by the John Hopkins University, Applied Physics Lab in satellite $1961\alpha\eta 2$, and later and more successfully in satellite 1963 22A. Passive techniques were used for initial satellite orientation and damping in a 400 n.mi.

MODIFIED INERTIA CONFIGURATION FOR PASSIVE ORIENTATION AND DAMPING



γ is the angle between the body X axis and the orbit plane (VZ)

δ is the angle between the damper rod and the orbit plane

β (not shown) is the angle between the damper rod and the XY plane

$(\delta + \gamma)$ is fixed at some constant value depending on inertia ratios $-15^\circ \leq \beta \leq 15^\circ$

FIGURE 3.1-1

circular orbit. Following insertion into orbit, a magnetic stabilization system was employed to despin the satellite and to orient the desired face toward the earth during passage through the north magnetic hemisphere. A 100 foot extensible boom was then deployed to increase the roll and pitch inertias and hence increase the roll and pitch restoring torques. Passive damping of satellite oscillations was provided by the spring mass system through the dissipation of energy by hysteresis loss in the spring. Results were largely successful although time to damp exceeded three weeks.

Tinling and Merrick (Reference 1) of Ames Research Center investigated the use of inertia coupling to provide passive three axes stabilization in gravity oriented satellites. Their configuration, shown in Figure 3.1-1 consists of a boom directed along the desired earth pointing axis and a movable rod coupled to the satellite through a damper mechanism. Coupling was created by locating the damper rod axis so that it and the principal axes of the satellite body, located in the horizontal plane, did not coincide and were both skewed to the orbital plane. A disturbance torque about any one axis resulted in relative motion between the satellite and auxiliary rod, which operated the damper. The function of the damper was to dissipate the energy of the satellite oscillations. Tinling and Merrick performed a theoretical analysis utilizing a digital computer simulation program on a limited number of configurations to find optimum system parameters. Analytical results presented in Reference 1 showed damping of satellite motions in a relatively short (less than 4 orbits).

The Advanced Technology Satellite (ATS) program, planned for late 1966, will include three axis gravity gradient orientation experiments on three of the five vehicles. The first satellite will employ four stabilization rods in an X configuration and a libration damping boom to counterbalance perturbations about all three axes of the spacecraft. The first test will be at high altitude, employing a 6500 n.m. circular orbit. There will be on-board capability to vary parameters such as magnetic dipole, center of solar pressure, length of the rods, and angle between the rods.

Since the ATS program will be confined to high altitude experiments and the past APL gravity gradient satellite stabilization program was only concerned with two axis orientation, there is a significant portion of the operational region for passively oriented vehicles that is as yet unexplored. This experiment is designed to fill in the information gap in the region of low altitude three axis stabilization by passive means.

PASSIVE SATELLITE ORIENTATION SYSTEM FUNCTIONAL BLOCK DIAGRAM

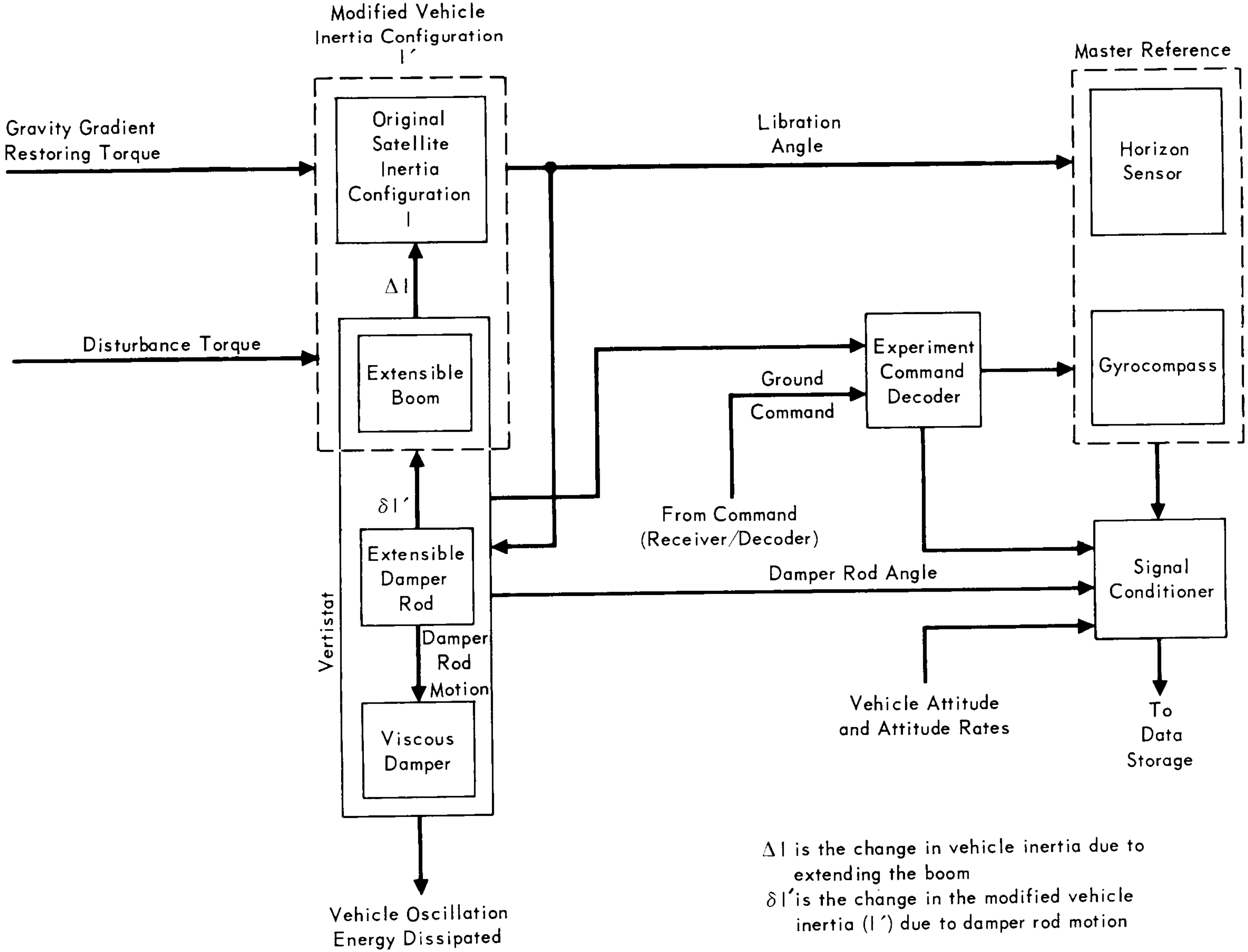
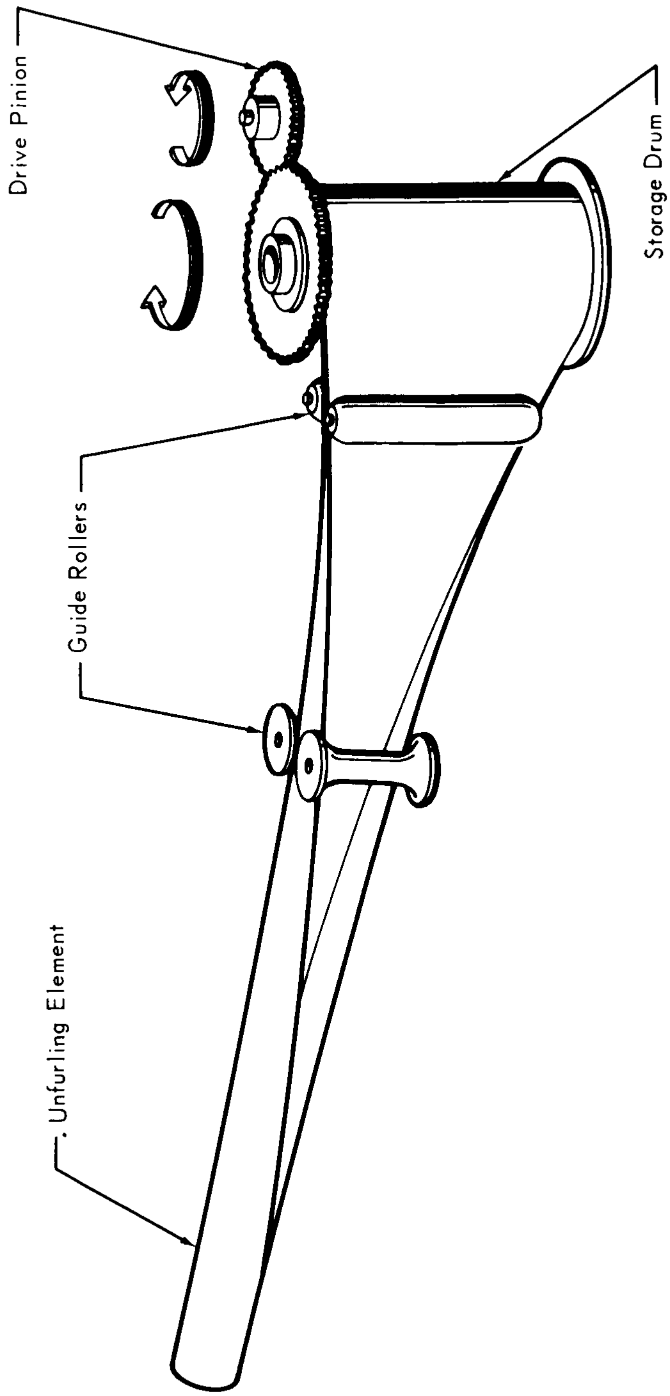


FIGURE 3.1–2

3.1.3 Functional Description - Figure 3.1-2 is a functional diagram of the augmented satellite and the supporting equipment required to perform the experiment. A compact unit called a Vertistat (see Reference 2), whose basic components are a damper mechanism and three spools of coiled metallic tape, is used to augment the satellite inertia configuration. Extension of the spools results in an inertia coupling configuration similar to that shown in Figure 3.1-1. When extended the tapes curl and form rods and the spools become tip masses. The function of the Vertistat is to increase the restoring torques tending to align the desired inertia axes to the earth local vertical and the orbit plane and to provide libration damping. Any mode of rotation or oscillation other than the desired one (one revolution per orbit) will provide differential torques on the damper tubes and primary tube. These differential torques will drive the viscous dampers and dissipate energy. Thus torques will always appear at the viscous dampers to oppose any undesired mode of motion.

Not shown in the functional diagram is a device needed to despin the satellite, assuming that a spin-stabilized carrier vehicle is used, or to invert the satellite should its initial stable orientation be upside down. The first requirement can be met with magnetic hysteresis rods or weights extended out from the satellite (yo-yo concept) which can reduce the spin rate to an acceptable level of 0.1 rad/sec. A one shot inertia wheel can be used to invert the satellite if required.

Experiment Equipment - A vertistat assembly contains three spools, one for the primary tube and two for the secondary or damping rods. These spools are mounted on brackets and are provided with squib actuated release latches. At the release signal, all three spools slowly unroll outward. A mechanical stop and



VERTISTAT TUBE

FIGURE 3.1-3

governor are provided on the spools to permit extension to half length and to control the rate of extension to minimize the effects of reaction torques. A small motor and gear drive are included to permit retracting the tapes as desired. The spools or tip masses are attached to the rods by explosive bolts and may be jettisoned upon command. The primary tube increases the pitch and roll inertias and the damper rods increase the yaw inertia. The damper rods are supported by a flexure pivot and are coupled to the satellite by a viscous damper.

The rod construction, illustrated in Figure 3.1-3, is a thin wall, rolled tape either of stainless steel, beryllium-copper, or fiberglass laminate. The rod is stowed flattened out and rolled up. When the roll is released, the elastic stresses in the material cause the strip to unroll away from the vehicle and to curl into the tubular form. (Note that this coiled strip is similar to the ordinary coiled steel measuring tape with the curvature extended to a full circle). The unrolling governor is pivoted at the center of the coil and carries a centrifugal brake. The construction and action are similar to that of the governor on a telephone dial.

The boom has one end attached to the spool on which it is rolled and the other end rigidly attached to the vehicle. The damper rod is formed by uncurling two identical spools at opposite ends of the damper mechanism. A pivot at the center is provided to permit limited damper motion.

Since the torques on the secondary, or damper, are of the magnitude of hundreds of dyne-centimeters, care must be taken to prevent pivot stiction. A flexure pivot is inherently a low stiction device. The pivots used in this application are helical torsion springs.

Experiment support equipment consists of a master attitude reference, such as a horizon sensor-gyrocompass combination, rate gyros to measure vehicle angular rates, pulsed jets or magnetometer coils providing attitude perturbations to test stability and damping, a command receiver to initiate tests, and a data handling unit to record time histories of experiment parameters and vehicle motion.

Test Method - Initial tumbling or spinning motion of the vehicle is stabilized by extending the Vertistat tubes when the angular rates have been reduced to 0.1 rad/sec. When stabilization is accomplished after tumbling, it may be with either end up. This ambiguity may be compensated with a one-shot inertia wheel. If an invert order exists, the wheel makes a fixed number of revolutions and stops. This inverts the satellite.

When the desired vehicle orientation in orbit is obtained, a series of experiments are conducted to evaluate the potential capability of the passive damping system. One set of experiments are performed by observing the free motion of the satellite. Detailed time histories of the large motions expected between the time of orbit injection and initial acquisition are stored for transmission. The small angle motion following acquisition is monitored for sufficient time to determine system damping.

Then, attitude perturbations are excited about each axis individually by controlled currents in a magnetometer coil or by small pulsed jets. Time histories for the resulting motion are obtained. The procedure is then repeated for changes in the inertia configuration caused by further boom extensions or possible retractions.

Additional experiments are intended to provide design information on the gravity gradient restoring torques required to limit disturbance torque errors to an acceptable level. The boom and damper rods will initially have tip masses and will be extended to full length. Capability to retract the rods and boom halfway or completely, and to jettison the tip masses will be provided in order to create additional inertia configurations for test. Time histories of the motion resulting from controlled disturbance torques will be stored and dumped upon command.

Error Sources - Significant error sources affecting the stabilization accuracy with respect to the earth local vertical include orbit eccentricity, solar radiation, atmospheric effects, and gyroscope effects. These, and a few relatively minor disturbance effects, are discussed in the following paragraphs.

Orbit eccentricity causes a periodic variation of the orbital angular velocity which acts as a driving frequency equal to the orbit frequency causing the satellite to oscillate. Reference (3) states that a dumbbell configuration will have about one degree pointing error for each 0.01 eccentricity.

Additional disturbance effects consist of gravitational anomalies, magnetic field interactions, solar radiation, atmospheric drag, and micrometeoroid impacts. Many do not contribute significantly to pointing error. For example, gravitational effects due to non-sphericity of the earth amount to a small fraction of a degree in pointing error. Alignment errors due to magnetic effects occur because of residual magnetic dipole in the satellite and are minimized at low altitude and high inclination orbits. This source of error can be reduced by careful demagnetization of the satellite. Micrometeorites impacting the vehicle, according to Reference 3, produce an average steady state error of about 0.5°. Center of mass shifts due to internal payload movement must be severly restricted. Manned vehicles or vehicles containing heavy movable observation equipment must be eliminated. Duty cycles of on-board equipment must not significantly couple into the stabilization system.

Solar radiation effects at low altitude can produce torques on the vehicle and an unsymmetrical loading of the extended rods causing bending due to differential thermal expansion. A small amplitude, high frequency (several cycles per minute) satellite oscillation can result unless boom oscillation damping is provided as well as proper surface treatment of the boom to reduce its response to thermal gradients. Disturbing torques on the vehicle due to solar pressure can be reduced by configuring the vehicle symmetrically about the center of mass.

Atmospheric effects at 400 n.mi. altitude can disturb the desired orientation of a gravity gradient satellite. Aerodynamic torques on the satellite are strongly dependent on vehicle shape. The magnitude of the torque due to drag is approximated by

$$\text{Torque} = C_D \, L_C \, Aq$$

Where C_D is the drag coefficient, L_C is the distance the center of area is offset from the center of gravity, A is the total area bombarded by the air molecules, and q is the dynamic pressure. For a particular satellite where C_D and A are constant, the designer must seek to minimize L_C to reduce aerodynamic disturbance torques.

Atmospheric rotation represents another source of aerodynamic disturbances. For a near earth polar orbit, the velocity relative to the rotating atmosphere is shown in Reference (4) to be 4° out of the orbit plane. This source of error can be reduced by restricting the experiment to low inclination orbits.

The rotating elements of gyroscopic sensors have an associated angular momentum vector ($\overline{H}$) directed along the spin axis. Torques will arise due to the time rate of change of $\overline{H}$ caused by vehicle angular rates and orbit rate. The former will be assumed to average to zero in one orbit. In the worst case orientation where $\overline{H}$ is perpendicular to the orbit angular rate vector, ω , the disturbance torque, T, is given by

$$\overline{T} = \overline{H} \times \overline{\omega} = |H| \cdot |\omega| \sin 90^\circ$$

For a typical gyro, H is 10^5 gm cm^2/sec. For a circular 400 n.mi. altitude orbit, ω is approximately .001 rad/sec. T is then 100 dyne-cm. The gravity gradient restoring torque on a 150 pound satellite with a 150 foot long boom ($I_{max} - I_{min} = 600$ slug-ft^2) is approximately 500 dyne-cm per degree of attitude error. The disturbance torque is seen to be significant. Therefore, gyros should not be used on the vehicle, or the gyro should be aligned so the gyro combination has no net momentum.

Uncertainties in the master reference measurements will limit the accuracy of the data measured. The horizon sensor-gyrocompass combination is expected to contribute an uncertainty of 0.5° to vehicle attitude measurements.

3.1.4 Experiment Physical Parameters - The physical dimensions required of the Vertistat depend on the shape, moments of inertia and mass of the vehicle, the orbital altitude, and desired pointing accuracy. The Vertistat will be required to modify the existing vehicle inertia configuration so that sufficient stabilizing torques will be available to provide the desired pointing accuracy. The stabilizing torques (T) expressed as a function of the moments of inertia (I) and distance from the center of the earth (R) are:

$$T_{pitch} \propto (I_{roll} - I_{yaw})/R^3$$

$$T_{roll} \propto (I_{pitch} - I_{yaw})/R^3$$

$$T_{yaw} \propto (I_{pitch} - I_{roll})/R^3$$

From the above equations the gravity restoring torque in a low altitude orbit (approximately 1 earth radius) is about 200 times that in synchronous orbit (approximately 6 earth radii). The Vertistat size requirements to obtain sufficient restoring torque for low altitudes will therefore be much less than at the higher altitudes. For a 1000 pound symmetrical vehicle in a circular 400 n.mi.

altitude low inclination orbit, a Vertistat system weight of 30 pounds comprised of a 100 ft long rigid extendable boom along the earth pointing axis and a 45 foot long damper boom skewed to the orbit plane will provide the desired pointing accuracy of 1 to 3 degrees. Yaw alignment of the vehicle depends strongly on the chosen angle in the horizontal plane between the damper rod and the original vehicle roll axis, and on their respective inertias. These parameters determine the orientation of the resultant principal axis of inertia. This axis will be acted upon by centrifugal torques tending to align it in the orbit plane. The experiment physical parameters summarized in Table 3.1-1 are based on the 1000 lb vehicle in 400 n.mi. orbit.

3.1.5 Data Parameters - Important experiment data parameters are given in Table 3.1-2. The basic data required includes, attitude reference information from the horizon sensor and gyrocompass, vehicle angular rates from the rate gyros, Vertistat configuration data, damper rod angle, indication of inverse

TABLE 3.1-1
PASSIVE GRAVITY GRADIENT
EXPERIMENT PHYSICAL PARAMETERS

ITEM	VOLUME		WEIGHT	POWER (AVG. WATTS)	
	(CU.FT.)	(L - W - H)	(LB.)	PEAK	NOMINAL
Experimental Devices Vertistat Assembly	1.2	Cylinder 1.5' Dia. x 0.5' height	32.0	0 (System is Mechanical)	
Support Equipment Signal Conditioning Electronics Power Supply	 .02 .02	 3 x 3 x 3 3 x 3 x 4	 1.0 2.0		 1.0 1.5
TOTALS	1.24		35.0		2.5

orientation, inertia wheel on - off function, thruster actuation signals, and data handling equipment functions.

3.1.6 Vehicle Orbit and Attitude Requirements. - A 400 n.mi near circular orbit having orbital eccentricity less than .01 is desirable for the test. Orbital inclination in excess of 30° increases sensitivity to aerodynamic effects of atmospheric rotation which can cause a pointing error in excess of 1°. Initial spin rate imparted to the vehicle for stability could result in an excessive bending moment on the extensible Vertistat members. Therefore some means of reducing this spin rate to perhaps 0.1 rad/sec (magnetic hysteresis rods might be used) must be employed prior to extending the Vertistat elements. Extending the rods and boom will then reduce the residual rates by increasing the moments of inertia and will also dissipate the remaining tumble momentum through action of the damper.

3.1.7 Experiment Support and Data Handling Requirements - The tests called out for this experiment do not impose excessive state-of-the-art requirements on support or data handling equipment. The on-board satellite three axis orientation reference will be provided by a horizon sensor for pitch and roll information and by a gyrocompass for yaw sensing. The reference will provide an angular accuracy of at least $\pm 0.5^{\circ}$ about all axes. Rate gyros will be employed to sense vehicle angular rates caused by thermal bending of extensible members during the test. The horizon sensor-gyrocompass master reference is discussed in Appendix A. The Vertistat must be mounted externally on the satellite with care taken to insure that the extended booms do not interfere with antennas, television lenses or other experiment equipment. The Vertistat is a mechanical device and requires only enough electrical power to control spool extension and retraction. The experiment sequence will be almost entirely commanded from the ground.

TABLE 3.1–2

PASSIVE GRAVITY GRADIENT EXPERIMENT DATA PARAMETERS

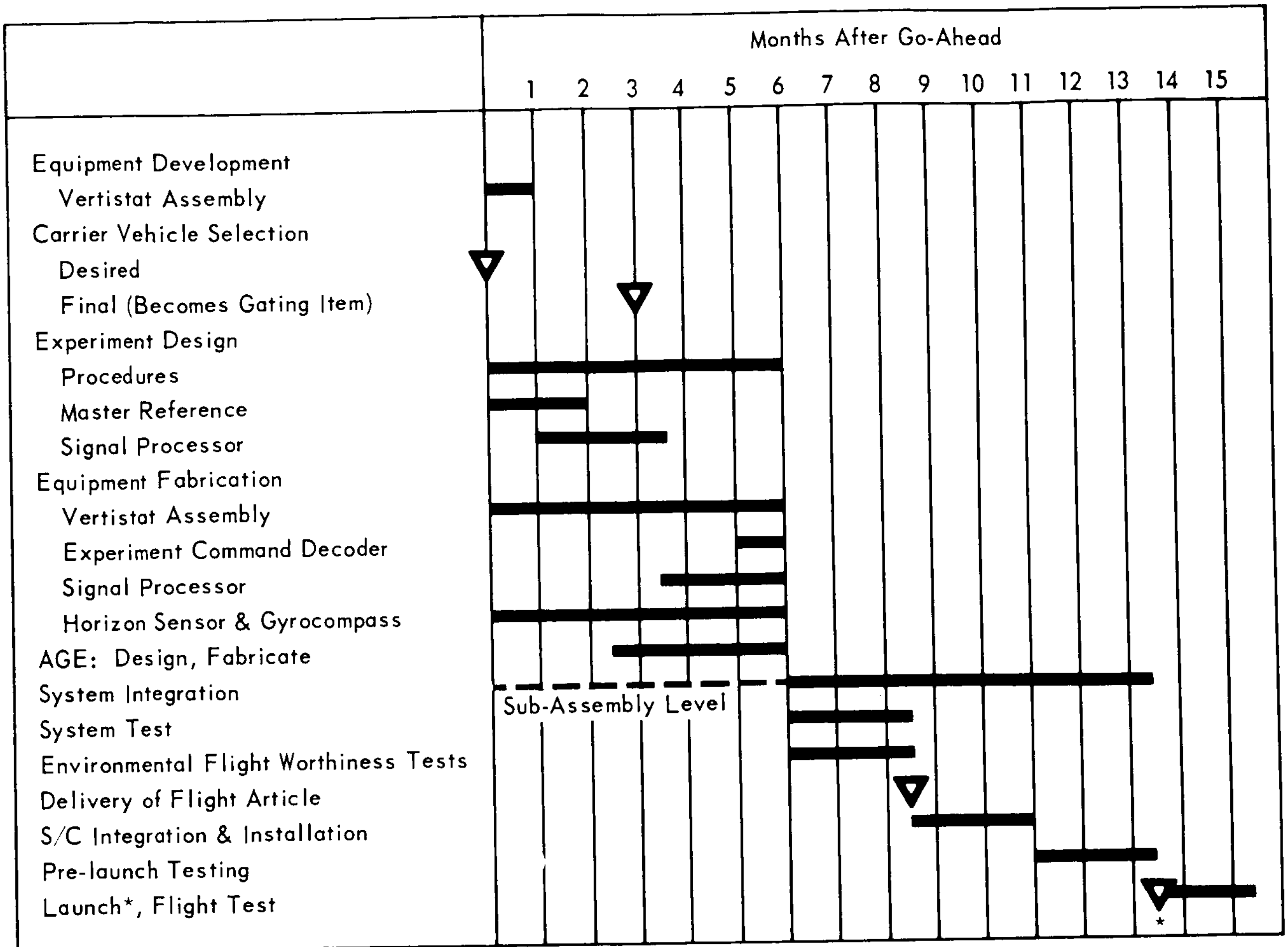
PARAMETER	SIGNAL FORM	RANGE OF PARAMETER	ACCURACY	SAMPLE RATE OR FREQ. RESPONSE	RECORD	ESSEN-TIAL	CON-TINUOUS
Horizon Sensor							
Pitch	Analog	± 20°	2%	sample once every 5 minutes	X	X	X
Roll	Analog	± 20°	2%		X	X	X
Gyro Compass	Analog	± 10°	2%		X	X	X
Rate Gyros	400 cps						
Yaw Rate	Analog	.05°/sec.	2%		X	X	X
Pitch Rate	Analog	to	2%		X	X	X
Roll Rate	Analog	40°/sec.	2%		X	X	X
Vertistat							
Boom Deployment	Digital	on-off			X	X	
Boom Length	Digital	on-off			X	X	
Damper Boom Angle	Analog	0 to 30°	2%		X	X	X
Right Side Up Indication	Digital	on-off			X	X	
Inertia Wheel on/off Function	Digital	on-off			X	X	
Thrusters No. 1, No. 2, No. 3 Actuation	Digital	on-off			X	X	
Back Up Timer-Boom Deployment Signal	Digital	on-off			X	X	

An experiment command decoder will be employed to minimize the interconnections between the command received decoder and the experiment. Data handling requirements for this experiment are basically those defined by Appendix B. Data will be in analog form with off-on functions superimposed on continuous channels. Recording durations of at least seven hours may be required. Signal sampling frequency will be on the order of 0.1 to 0.01 cps.

3.1.8 Experiment Flight Test Plan - Figure 3.1-4 shows the experiment development plan. The schedule is an estimate based on the required development, integration testing and flight time and is included for planning purposes. The major gating item in the schedule is integration of the Vertistat package with the vehicle chosen. Failure to select a carrier vehicle early in the program would also delay the program.

Ten commands are required from the ground to perform the test sequence shown in Figure 3.1-5. The test sequence begins when vehicle angular rates have been reduced to 0.1 rad/sec. A command is given to apply power to the master reference. Following this, commands are given to start recording data parameters and to deploy the Vertistat tapes. Commands to dump data are given at opportune times when the site and satellite are in coincidence. Commands to actuate pitch, roll, and yaw axis thrusters are transmitted separately when transients due to each have subsided and data has been dumped. The sequence is repeated following a command to extend the Vertistat tapes to full length. Both sequences are repeated following jettisoning of tip masses. This last test may of necessity be omitted or delayed, if future passive vehicle three axis stabilization is required since the resulting inertia ratios may not provide sufficient stabilizing torque capability.

PASSIVE SATELLITE STABILIZATION EXPERIMENT DEVELOPMENT PLAN



*Depends on Carrier Vehicle – may vary from 1 month to 8 months after installation.

FIGURE 3.1-4

Post experiment data reduction is performed by determining damping characteristics from time histories of satellite motion following disturbance inputs for the inertia configuration tested.

It should be recognized that the preceeding experiment description is based upon one chosen configuration. The approach of Tinling and Merrick was selected because it was technically feasible and promised to minimize weight and volume by using only one damper rod. Kamm's Vertistat, which uses a viscous damper, was

GROUND COMMAND SEQUENCE – PASSIVE STABILIZATION

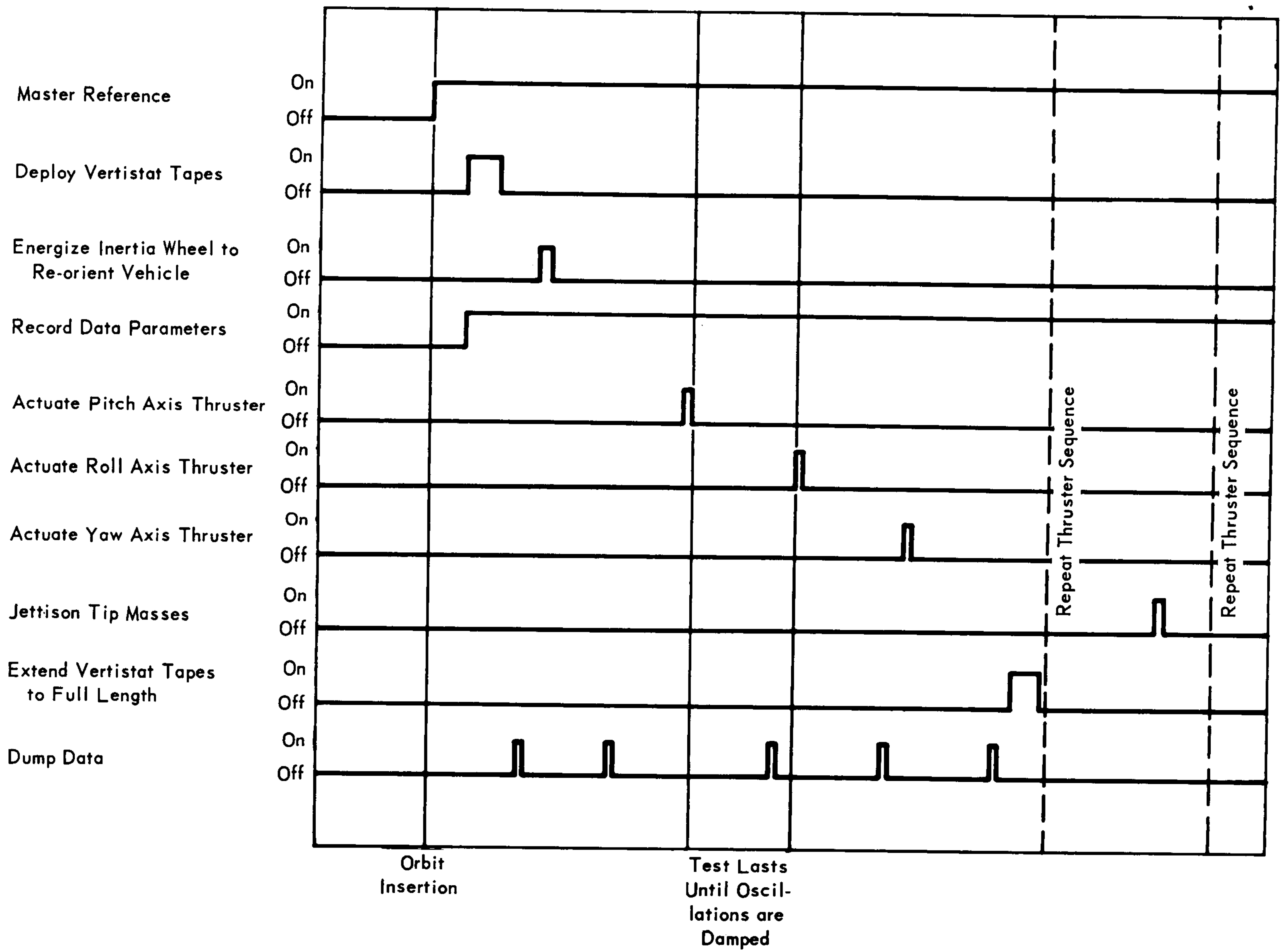


FIGURE 3.1-5

selected because it was light and compact and the viscous damper physical properties could be varied sufficiently to provide a wide range of damping coefficients. The flexure pivot eliminated dead band to a great extent, enabling the damper to respond to small amplitude motion. The ATS program may very well provide data that would influence the choice of system components for this low altitude test. If so, there will be no problem in incorporating late modifications.

REFERENCES

1. Tinling, Bruce E. and Merrick, Vernon K., Exploitation of Inertia Coupling in Passive Gravity-Gradient-Stabilized Satellites, Journal of Spacecraft and Rockets, Vol. 1, No. 4, July-August 1964.

2. Kamm, Lawrence J., "Vertistat": An Improved Satellite Orientation Device, ARS Journal, June 1962.

3. Katucki, Richard J., Gravity Gradient Stabilization, Space/Aeronautics, October 1964.

4. Schrello, D. M., Aerodynamic Influences on Satellite Librations, ARS Journal, March 1961.

3.2 Attitude Determination by Ion Sensing

3.2.1 Objective - The primary purpose of the experiment is to demonstrate the feasibility of determining vehicle yaw attitude for near earth orbital vehicles by ion sensing techniques. It is further desired to obtain a measure of the attitude sensing accuracy. Finally, it is desired to instrument the experiment to obtain a partial evaluation of the effects of vehicle-ionosphere interaction on ion attitude sensing.

3.2.2 Background - For near earth orbital vehicles, the sensing and subsequent control of vehicle yaw relative to the orbit plane is essential for detailed earth observations or for re-entry after an appreciable time in orbit. Space vehicles which have required or will require yaw control include Discoverer, Mercury, Nimbus, OGO, Gemini and Biosatellite. On these programs the yaw reference signal for control is obtained from some form of "gyrocompass" (see Paragraph 3.3) which operates in conjunction with a horizon sensor. It will be desirable to have a coarse yaw reference to serve as an acquisition aid for precise gyrocompassing and also as a reliable backup in the event of primary system failure. Presently the coarse reference for initial gyrocompass acquisition is obtained from initial conditions (attitude at injection), solar aspect sensors or man's yaw sensing capability. Several of the programs have no reliable backup to operate in the event of gyrocompass or horizon sensor malfunction. Hence, it is considered desirable to conduct orbital experiments to develop alternate techniques for sensing yaw. Various techniques have been considered such as celestial rate, V/H sensing, magnetometer, solar eyes, sub-satellite plus others. The ion sensing method has been selected for an early orbital experiment because the ion probe is relatively simple and parts of the device have been flown. The physical phenomenon, upon which ion sensing is based, is described in the following paragraphs.

From an altitude of about 60 nautical miles to several thousand nautical miles, the earth's atmosphere is ionized. The motion of the ions is believed to be governed primarily by electrodynamic laws, so that the ionization "cloud" which surrounds the earth remains fixed with respect to magnetic coordinates and therefore, rotates with the earth. The ionization to be used for sensing yaw can be considered a Maxwellian gas with an average energy of less than one electron volt.

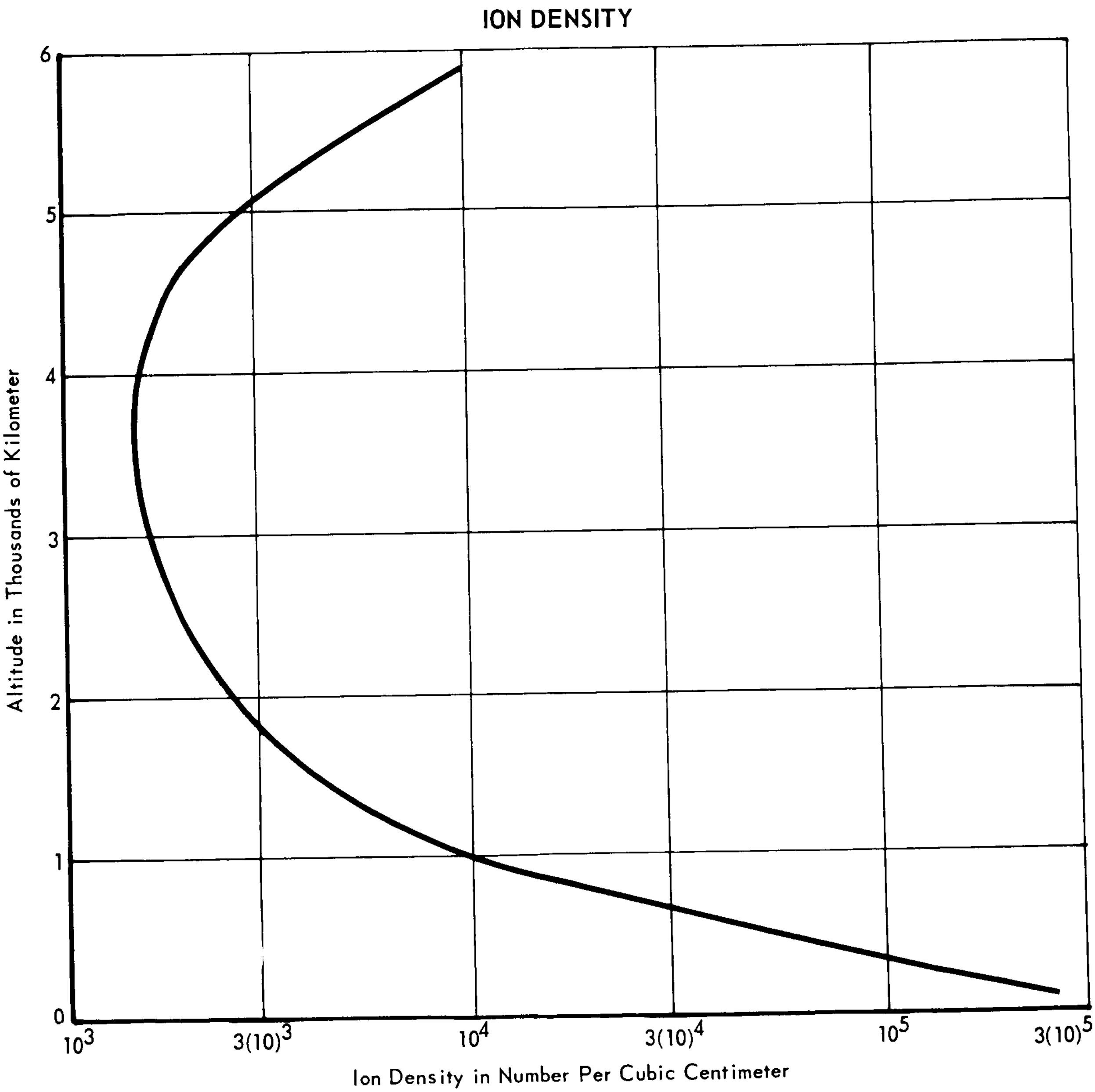


FIGURE 3.2-1

These ion particles should not be confused with the highly energetic protons and electrons of the Van Allen Belts (Reference 1).

An ion probe was successfully flown by the Los Alamos Scientific Laboratory on a Scout rocket from Wallops Island to more than 3,300 miles altitude. The data has been analyzed at North Carolina State University. That work demonstrated that the ions in the upper atmosphere are more than 100 times as dense as needed for attitude sensing by ion probe methods up to much more than 3,300 miles and probably to beyond 25,000 miles. The observed densities are shown in Figure 3.2-1 (Reference 2).

It is possible to sense pitch and yaw attitude relative to the velocity direction as the vehicle moves through the ion cloud. However, it should be recognized that vehicle pitch and yaw attitude determined by ion sensing are not the same as vehicle pitch and yaw determined by horizon sensor - gyrocompassing techniques except in special cases.

Ion sensed pitch and horizon sensed pitch coincide continuously only in circular orbits where the velocity vector is perpendicular to the local vertical. For non-circular orbits, coincidence occurs only at apogee and perigee, and the maximum deviation occurs at the minor axis crossings (about 1 degree maximum deviation for an eccentricity of 0.02).

Gyrocompassed yaw and ion sensed yaw coincide continuously only on equatorial orbits where ion cloud velocity and vehicle inertial velocity directions are colinear. For inclined orbits, coincidence occurs only at the maximum North or South latitudes with maximum deviation occurring at equatorial crossings. The magnitude of the maximum deviation is a function of orbit inclination and altitude, typically not more than 2 or 3 degrees for a polar, near earth orbit.

For many missions, control to either attitude reference would provide adequate accuracy. Other missions may require precise control to either the orbit plane-local vertical reference or to the relative velocity reference. As implied in the previous paragraphs, computed compensation from one reference to the other is relatively simple since the deviations are sinusoidal in the true anomaly angle. However, navigation data is required onboard to compute the compensation continuously.

ION PROBE PAIR

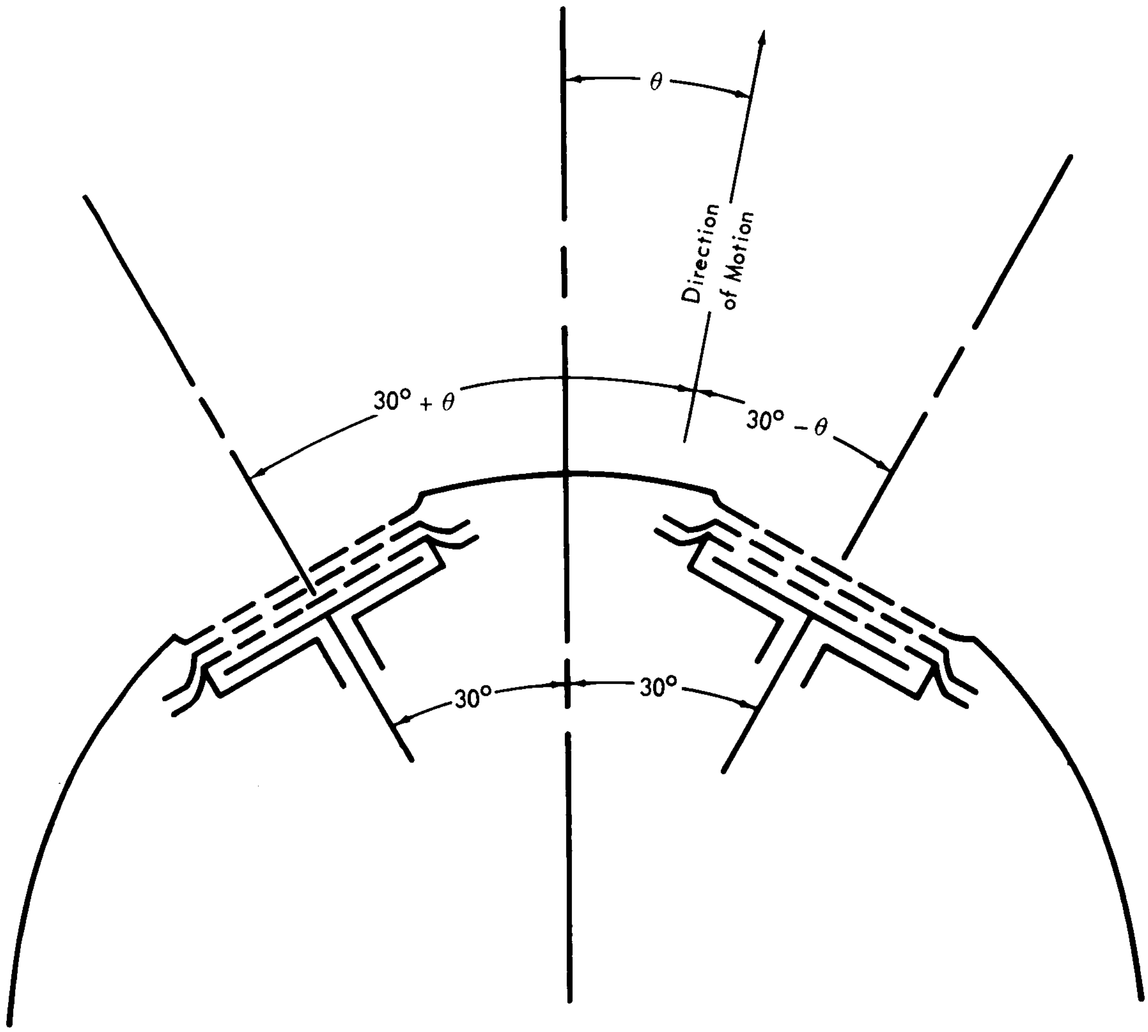


FIGURE 3.2-2

. . Several ion sensing devices can be configured which provide yaw and pitch information. Reference 1 suggests several methods for sensing yaw including:

a. Phase detector,

b. Shadow cup, and

c. Oscillating cup.

All of these methods use ion probes that are essentially open ended vacuum tubes, which, at orbital velocities, sweep up thermal ions.

Figure 3.2-2, taken from Reference 2, shows schematically how a pair of ion probes might be arranged to provide one axis (pitch or yaw) of attitude information. The device shown in the figure is a shadow cup configuration in which the normals to the electrodes are set at an angle of 30° from the vehicle axis. If the axis of the vehicle inclines in the plane of the probes to an angle θ from the direction of motion, and if the axis of the vehicle also inclines in the other direction simultaneously to an angle Φ, the resulting ion currents when subtracted yield a resultant potential which is proportional to the logarithm of the ratio of the two currents. Thus, the output directly measures the angle of inclination θ independently of: (1) density of ions in that part of the atmosphere; (2) the velocity of the vehicle; or (3) the angle Φ at which the probe pair is tilted in the other direction from θ.

An ion sensor experiment is planned for an early Gemini low inclination orbit mission. The sensor for the Gemini test consists of two ion cups mounted on a short boom to indicate error in pitch and yaw from the direction of motion. Principle results will be obtained from measurements of the difference in signal level between the two cups. A more inclusive experiment is proposed for this orbital test program. A polar orbit will be used to determine ion density variations as a function of latitude. Two identical sensors will be used, one mounted to the

ION ATTITUDE SENSING EXPERIMENT FUNCTIONAL BLOCK DIAGRAM

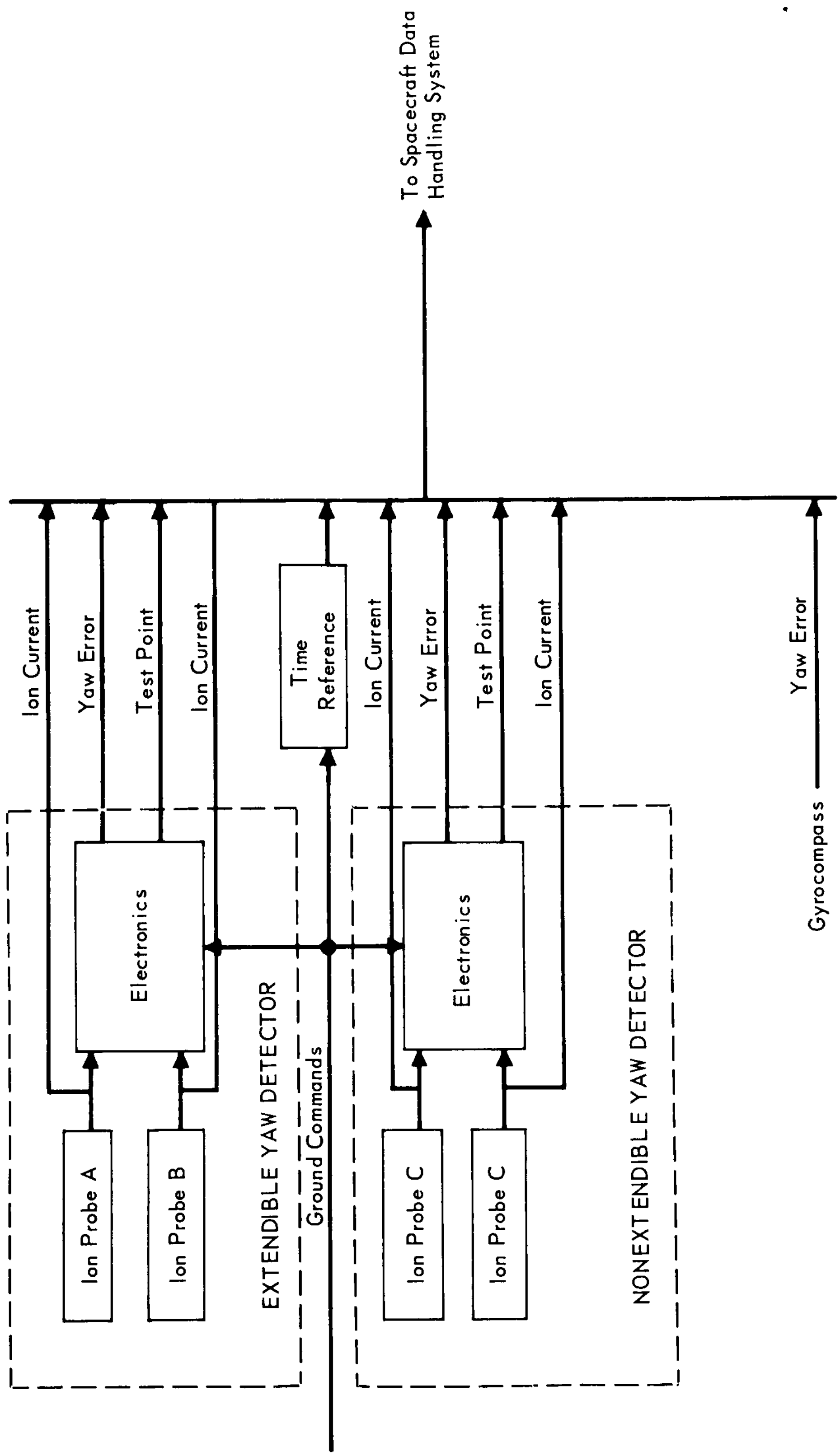


FIGURE 3.2–3

vehicle and one extended on a short boom. Comparison of their outputs will provide data on interaction of the ion field with the vehicle. A long mission duration is planned to adequately define ion field variation due to solar activity, day-night variations, etc.

3.2.3 Function Description - A shadow cup type of device is assumed as the basic configuration. It is desired to sense vehicle yaw only, i.e., not pitch and yaw. Figure 3.2-3 provides a functional description of the experiment. Two sets of detectors are used which are identical as to operation and orientation to the vehicle yaw axis. However, one detector set is mounted on an extendable boom to isolate as much as possible the effects of vehicle interaction with the ion cloud. Each detector contains two ion probes and appropriate electronics. The active areas of the probes form angles of approximately $\pm$ 30° with the vehicle axis.

Experiment Equipment - Two ion yaw sensing detector systems are included in the experiment. Each detector system includes two ion probes plus electronics. The following paragraph on ion probe construction and operation has been abstracted from an Aero Geo Astro Corporation brochure supplied to McDonnell (Reference 2).

"The ion probe consists of three knitted grids and a collector plate. The knitted grids are the same as in the Bennett mass spectrometer which has been the most widely used instrument in measurements of the composition of the upper atmosphere, ionized and un-ionized, during the more than ten years that such measurements have been made. Figure 3.2-2 shows the arrangement of the electrodes in each of the ion probes. The outer grid is attached to the skin of the vehicle and serves to prevent electric fields caused by potentials applied to the inner grids from extending out into the surrounding atmosphere and distorting the ion flow. The middle grid is traversed between vehicle ground and a positive twenty volts at a frequency in the intermediate audio range selected to avoid cross-talk or any other interference from other equipment in the vehicle. The inner grid is held at a negative twenty volts to turn back all incoming electrons and to suppress all photo and secondary emission from the collector plate."

Test Method - The vehicle is controlled to a local vertical-orbit plane orientation by using a horizon sensor-gyrocompass attitude reference system. The gyrocompass serves as the master yaw reference system. Rough alignment of vehicle pitch and roll is required to prevent vehicle pitch from appearing as a yaw misalignment at the ion sensor outputs. When the vehicle axis is aligned in the plane containing the vehicle velocity vector, equal currents are measured in each probe, and zero yaw attitude error is indicated. A yaw error results in unequal probe currents, and the ratio of these currents is a measure of the yaw attitude error. After proper conditioning by the electronics, this current ratio is recorded for later telemetry to the ground. Also, the current from each probe is recorded to permit ground determination of the ion current density in each probe and ground computation of the yaw error. The yaw attitude error indicated by the gyrocompass master reference is also recorded to obtain a measure of ion attitude sensor accuracy.

Mission duration is not critical; however, it is desirable to conduct the experiment for several days, and, if possible, during periods of solar activity. Data is collected throughout each orbit to permit evaluation of the effects of day-night variation and possibly solar orientation variations. Data on the effects of vehicle shape and charge developed will be collected by placing an identical set of sensors on an extendable boom and comparing the outputs.

Major Error Sources - Since the ions are charged, they are subject to electrodynamic forces caused by space vehicle-ionosphere interaction. This interaction is a function of magnetic field, vehicle shape and charge developed, solar conditions and activity plus many other factors which make a meaningful prediction of these effects almost impossible. Mounting the probe pair on an extendable boom should minimize this source of error. Special care must be taken in boom design to minimize structural misalignments between the two sets of sensors particularly under the day-night variation.

Instrument measurement uncertainties are also a major error source. Some preliminary calculations of a probe pair of this general class, performed at North Carolina State University, have shown that this instrument will have an angular accuracy of less than 2 degrees for inclinations up to 25 degrees in either pitch or yaw or both simultaneously. It is further estimated that angular errors of less than 1/2 degree should be readily obtainable.

3.2.4 Experiment Physical Parameters - Experiment weight, volume and power requirements are listed in Table 3.2-1. Each detector system is estimated to weigh ten pounds. The shape is cylindrical with dimensions of six inches in diameter and nine inches long. The system is designed to operate from 28 VDC.

TABLE 3.2-1

ION ATTITUDE SENSING
EXPERIMENT PHYSICAL PARAMETERS

SYSTEM COMPONENT	SIZE		WEIGHT (LB.)	POWER	
	VOLUME ($FT.^3$)	L – W – H (IN.)		PEAK (WATTS)	NOMINAL (WATTS)
Experimental Device					
(2) Ion Detectors	0.28	6 dia. x 9	20		20
Electronics	0.08	3 x 6 x 8	4		
Support Equipment					
Time Reference	.01	3 x 3 x 2	0.5		1.0
Total	0.37		24.5		21.0

TABLE 3.2-2

ION ATTITUDE SENSING EXPERIMENT DATA PARAMETERS

DATA POINT	SIGNAL FORMAT	PARAMETER RANGE	SAMPLE RATE (BANDWIDTH)	ACCURACY	RECORD	ESSEN-TIAL	REAL TIME
Ion Yaw Error (2)	Analog	0–5 VDC	1/sec.	1%	X	X	X
Gyro Yaw Error	Analog		1/sec.	1%	X	X	X
Ion Current (4)	Analog	0–5 VDC	1/sec.	1%	X	X	X
Probe Temperature (4)	Analog	0–5 VDC	1/sec.	5%	X	X	X
Power to Probes (2)	Analog	0–5 BDC	1/sec.	20%	X	X	X
Timer	Digital		1/sec.	1 sec.	X	X	X

3.2.5 Data Parameters - Experiment data parameters are given in Table 3.2-2 in their order of priority. The primary information desired is a measure of attitude sensing accuracy of the ion sensor.

3.2.6 Vehicle Orbit and Attitude Requirements - Theoretically the experiment is not overly sensitive to altitude and eccentricity within bounds since the ion density variation is only about two orders of magnitude over the altitude range from 100 to 2,100 nautical miles and is less than an order of magnitude from 600 to 3,300 nautical miles (see Figure 3.2-1). However, the figure does not show the variation in current density which is a function of both ion velocity and density. It is the current density variation which effects the dynamic range of the sensors. In the interest of practical sensor design, it may be desirable to operate near the region of constant ion density in Figure 3.2-1. Ideally, the smallest density variation occurs around 2,100 nautical miles where a ± 300 nautical mile altitude variation might be permissible. If this same density variation ground rule is applied at the 300 nautical mile region, much smaller altitude variations are permissible, e.g., around ± 20 nautical miles. Additional study and vendor discussions are needed to determine if the experiment is sensitive to eccentricity. For the present, it is assumed that a near circular orbit is used. A near polar or high inclination orbit is desirable so that ion density variations and sensor performance as a function of latitude can be determined.

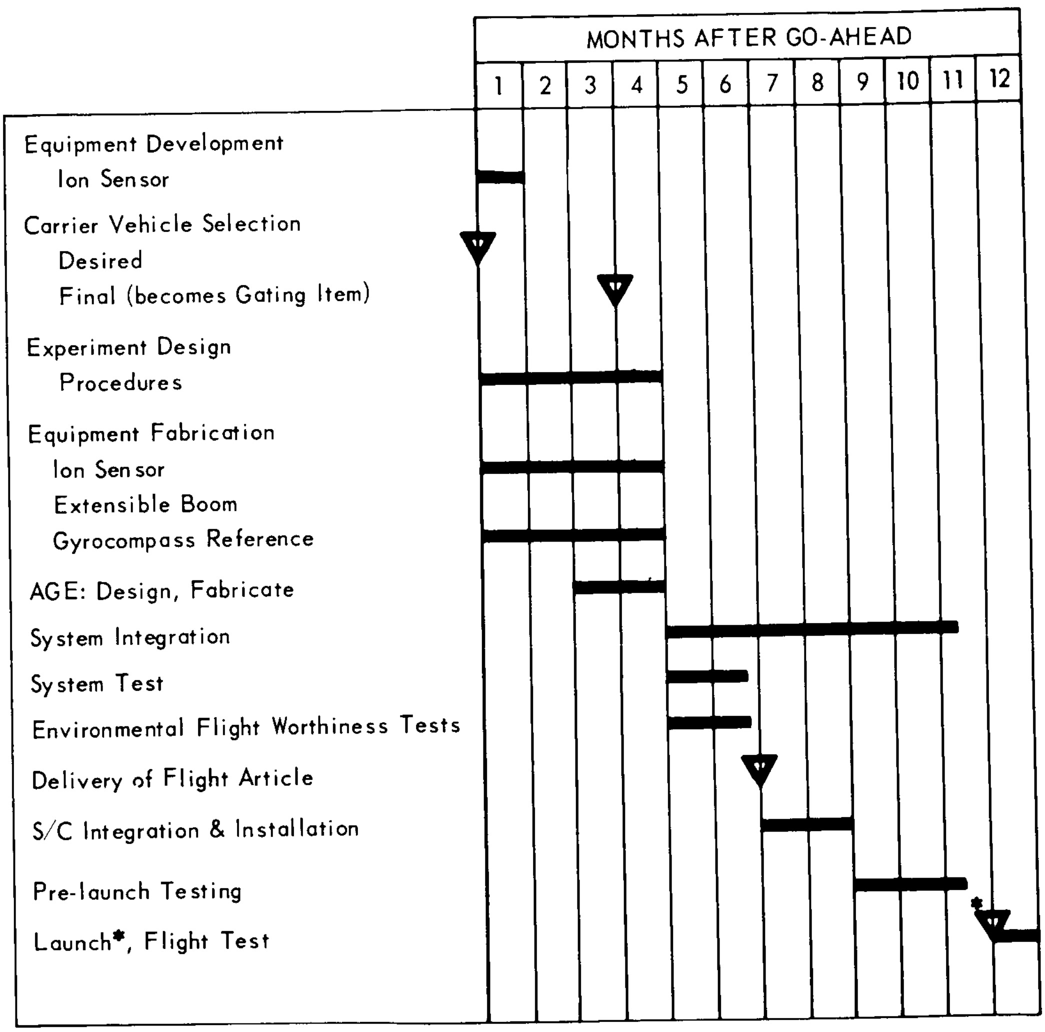
As stated previously, the preferred vehicle control is to an earth-orbit plane orientation. Attitude stabilization relative to this orientation is not critical. Only rough ($\pm$ 15°) pitch alignment is required. Roll control should be held to $\pm$ 3° or so to prevent vehicle pitch from appearing as a yaw misalignment at the ion sensor output. Vehicle yaw control requirements are similar to pitch. It would be desirable to permit the vehicle to yaw through rather wide limits ($\pm$ 25°) to determine the sensor performance under these conditions.

Similarly, the rate requirements are not severe. Permissible rates are determined by the sensor time constant and the attitude measuring accuracy. Assuming a 0.5 degree accuracy and a 2 second time constant, rates of 0.25 degrees per second are permissible.

3.2.7 Experiment Support and Data Handling Requirements - Signal conditioning is not required since all experiment signals are in the format required by the data handling equipment. The master yaw attitude reference for the test will be a precision gyrocompass accurate to within $\pm$ 0.5°. Discussion on the use of a gyrocompass as a master reference is contained in Appendix A. The "on-board" ion sensor must be mounted so that its field of view is not blocked. Provision must also be made to mount an identical ion sensor on an extensible boom outside of the area of vehicle-ion field interaction. Alignment between the centerlines of the two sets of sensors must be within $\pm$ 0.5 degrees. Assuming continuous operation for one week (168 hours), 7,200 watt-hours will be required for the mission. Two ground commands are required. One command is needed to start the experiment and a second is needed to dump data and reset the recorder. General data handling considerations are discussed in Appendix B.

3.2.8 <u>Experiment Flight Test Plan</u> - The proposed experiment development plan is shown in Figure 3.2-4. Design and fabrication of the experiment equipment should be complete in four months. Two months of system and flight worthiness tests will be required prior to delivery of the flight article. Four additional months

ION ATTITUDE SENSING
EXPERIMENT DEVELOPMENT PLAN



* Depends on Carrier Vehicle – may vary from 1 month to 8 months after installation.

FIGURE 3.2-4

are allowed for spacecraft integration and pre-launch testing.

The flight test sequence following the ground command to start test involves only the storage of data and data dump. The experiment variables deal only with changing environmental conditions and location. No programmer is required for the test.

Post-experiment data reduction will correlate time histories of the measured yaw attitude by both external and internal ion sensors with the reference yaw attitude. Qualitative data concerning vehicle-ion field interaction will be obtained by comparing the measurements of the two sensors.

REFERENCES

1. Ogletree, E. G., Interim Technical Documentary Report for Contract AF04(695)-289, MIT-Instrumentation Laboratory, Report No. R-422, dated 10 September 1963

2. "Technical Proposal to Provide an Ion Probe Attitude Sensor for Spacecraft", Aero Geo Astro Corporation - Space Engineering Laboratory

3.3 Gyrocompassing

3.3.1 Objective - The purpose of this experiment is to evaluate the accuracy of a precision gyrocompass mounted in an orbiting vehicle. Furthermore, this experiment will evaluate the feasibility of using a gyrocompass for an autonomous navigation attitude reference, or as a master reference for evaluating various other yaw sensing techniques.

3.3.2 Background - The determination (and subsequent control) of vehicle yaw orientation with respect to the orbital plane is important for missions which require earth observations or reentry after an appreciable time in orbit. While the vehicle can be controlled to the orbit plane yaw alignment from an inertially-aligned attitude reference and knowledge of orbit parameters, this approach requires a continuous on-board navigation computer capability and vehicle control to a computed reference. For long-term earth orbital missions it is desirable to provide an orbital plane yaw-reference sensor which operates independent of navigation data. The only known sensor which can provide a true orbit plane reference without recourse to navigation data is some form of a gyrocompass. Various gyrocompass configurations have been used or are planned for use in our space program as shown in Table 3.3-1. The basic input to all these configurations is the local vertical reference as sensed by a horizon scanner. Thus, the accuracy of the gyrocompass is a function of the horizon sensor and gyro performance as well as the vehicle attitude control system operation. With known signal and performance characteristics of the horizon sensor, gyro, and attitude control system, gyrocompass performance can be predicted by analysis, e.g., see References 1, 2 and 3. Furthermore, a laboratory rate table gyrocompass test can simulate orbital motion, attitude control operation, and horizon sensor performance. However, the horizon sensor performance uncertainties due to an imperfectly simulated earth (described in paragraphs 2.4 and 2.5) as well as the gyro drift uncertainty in the near zero-g environment limit

TABLE 3.3-1

GYROCOMPASS CONFIGURATIONS

VEHICLE	CONFIGURATION	REFERENCE
Discoverer	Strapped-down inertial reference, 3 rate integrating gyros, gyros operate in rate made for gyrocompassing.	1
Mercury	Two 2-degree-of-freedom gyros strapped to vehicle, one vertical gyro and one directional gyro.	2
Nimbus	Roll rate gyro strapped to vehicle.	3
OGO	Rate-integrating gyro strapped to vehicle.	4
Gemini	4-gimballed inertial platform.	5

the usefulness of both the analytical and lab test results. A further consideration of importance to this experiment is that apparently only the Nimbus vehicle (of those listed in Table 3.3-1) includes a master reference for evaluating gyrocompassing accuracy (Reference 4). According to the reference on pp 77-78, "The system's fine sun sensor is used to check drift in the yaw gyro..... If drift is discovered, ground stations will command a bias to be introduced into the summing amplifier to maintain accuracy of the yaw system." The master reference system for evaluating this precise gyrocompass experiment might be the same as or similar to the approach described above for Nimbus.

Each of the configurations in Table 3.3-1 has certain advantages which are closely related to the program on which the gyrocompass is used. The Mercury vehicle used conventional ball-bearing two-degrees-of-freedom gyros since this hardware was readily available and could be adapted quickly. Roll-rate gyros have been used on several programs - usually a rate-integrating gyro with its input axis aligned approximately along the vehicle roll axis and operating with the loop closed to perform gyrocompassing. The rate-integrating gyro configuration has several system advantages including:

a. small size,

b. flexibility of operation, i.e., can operate either as a rate or attitude gyro and used for gyrocompassing or attitude memory.

However, from analysis in Reference 3, it was shown that gyrocompassing performance with a strapped two-degree-of-freedom gyro was superior to that with a roll-rate gyro when each was used in the same vehicle control loop as a yaw nulling device. According to Reference 3, "The difficulty with the roll-rate gyro system is that it produces primarily a roll-rate signal in combination with a yaw-position signal caused by coupling. The presence of the large roll-rate signal from the roll-rate gyro greatly decreases the performance of this system. The two-degrees-of-freedom gyro system, however, essentially nullifies the effect of the roll-rate term by converting roll position to roll rate where it cancels the roll-rate signal from the gyro." The gimballed inertial platform will provide better gyrocompassing accuracy than the strapped configurations because spacecraft motion and horizon sensor noise can be filtered and partially isolated from the gyrocompass by the platform gimbal stabilization loops. Also, gyro drift is less significant since higher performance inertial-grade gyros are used. However, from the system test viewpoint, the inertial platform approach is not attractive because of design complexity (reliability is low) and the platform size, weight, and power are not competitive with the other two approaches. Design factors which make the inertial platform more complex include:

a. a 4-gimballed unit is used, and

b. accelerometers are mounted on the inner element.

GYROCOMPASS EXPERIMENT FUNCTIONAL BLOCK DIAGRAM

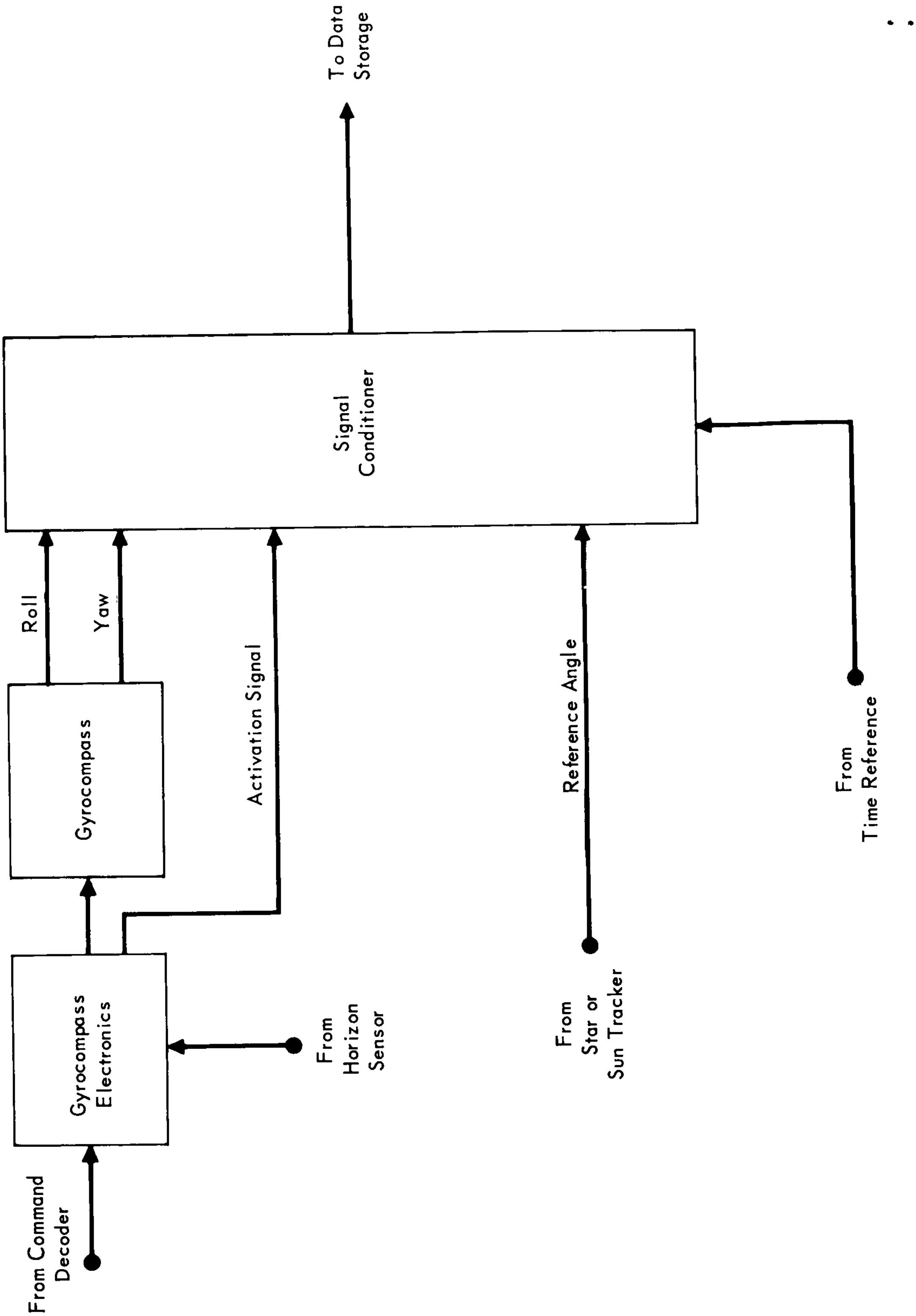


FIGURE 3.3-1

Depending on the attitude reference and guidance requirements during other mission phases, it may be possible to simplify the platform gyrocompass by using a 1-, 2-, or 3- gimballed configuration with accelerometers removed from the inner element. In fact, the Sperry Gyroscope Company (Reference 5) has designed a one-gimballed unit (company sponsored) and the MIT Instrumentation Laboratory (Reference 6) has designed and delivered a two-gimballed unit (USAF sponsored). The aim in both of these designs is to provide a yaw reference which is more precise than the strapped configuration and is less complex than a full inertial platform.

To permit definition of a gyrocompass experiment the following assumptions are made:

a. A precision gyrocompass (0.1 degree accuracy) is needed which can be used in the later experiments as a master reference to evaluate various yaw sensing techniques.

b. The precision gyrocompass should be as simple as possible to permit incorporation as a piggyback experiment on an existing vehicle.

3.3.3 <u>Function Description</u> - The gyrocompass and its supporting electronics, a master optical reference, and a data instrumentation system are shown in Figure 3.3-1. The gyrocompass configuration is a 2-gimballed platform stabilized by two single degree-of-freedom, rate-integrating, inertial-grade, gyros which are mounted on the inner gimbal. The gyrocompass platform has limited freedom in roll and yaw. The roll error signal input to the gyrocompass is provided by a horizon sensor. (The horizon sensor is assumed to be spacecraft equipment external to the experiment). The gyrocompass output signals indicate vehicle roll and yaw attitude errors relative to a local vertical/orbit plane coordinate system. The master yaw reference includes several solar aspect sensors whose output signals are used to compute (on the ground) vehicle yaw relative to the orbit plane.

FUNCTIONAL BLOCK DIAGRAM OF "COMPASS" PLATFORM-TYPE YAW SENSOR

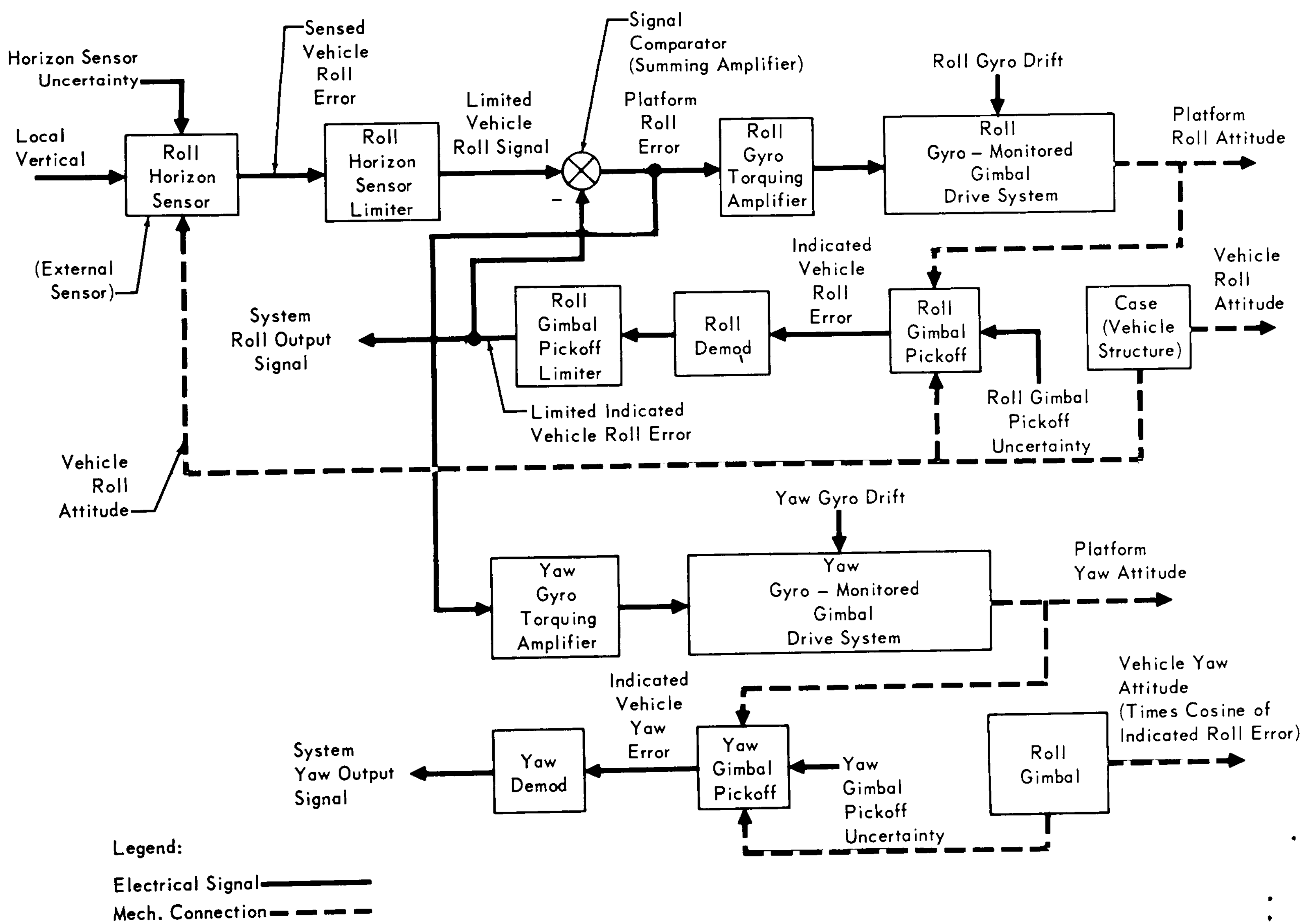


FIGURE 3.3–2

. . The functional block diagram of the gyrocompass control loop is shown in Figure 3.3-2. The system operates by comparing the sensed vehicle roll error out of the horizon sensor with the platform roll attitude to form a roll error signal. This signal is used to drive the platform roll gimbal, and is also crossfed to the yaw gyro torquing amplifier at a gain multiple of five. This results in the yaw attitude gimbal being slewed more rapidly to the null position than the roll gimbal thus reducing undesirable roll rate due to yaw error into the gyro.

Experiment Equipment - The gyrocompass equipment includes the instrument and an electronics package. The instrument, shown schematically in Figure 3.3-3, is a two-gimballed (roll-yaw) platform with two rate-integrating (IRIG) gyros mounted on the inner element. The inner gimbal represents vehicle yaw and each gimbal has $\pm$ 15 degrees of mechanical freedom. The gyros sense roll and yaw platform orientation relative to inertial space and generate error signals that drive the gimbal torquers when the platform roll or yaw axes deviate from the orbit plane. D.C. Pancake torquers are used and, since the gimbals have limited rotational freedom, brushes are not required. The gimbal positions are indicated by pancake synchro transmitters which have a maximum error spread of 6 minutes each. The electronics package includes the gyrocompass power supply and platform servo loop functions. The power supply accepts 28 VDC from a vehicle bus. Table 3.3-2 summarizes the equipment size, weight and power estimates. (These estimates are based on Reference 6 and discussions with MIT). Presently, the unit is designed to operate in an uncontrolled environment, but not exposed to direct sunlight. However, to assure proper gyro temperature control, the environmental temperature range must be known in advance.

An alternate simplified gyrocompass, described in Reference 5 and labeled the Orbital Vehicle Heading Indicator (OVHI) by Sperry, is shown schematically in

"COMPASS" YAW SENSOR SCHEMATIC

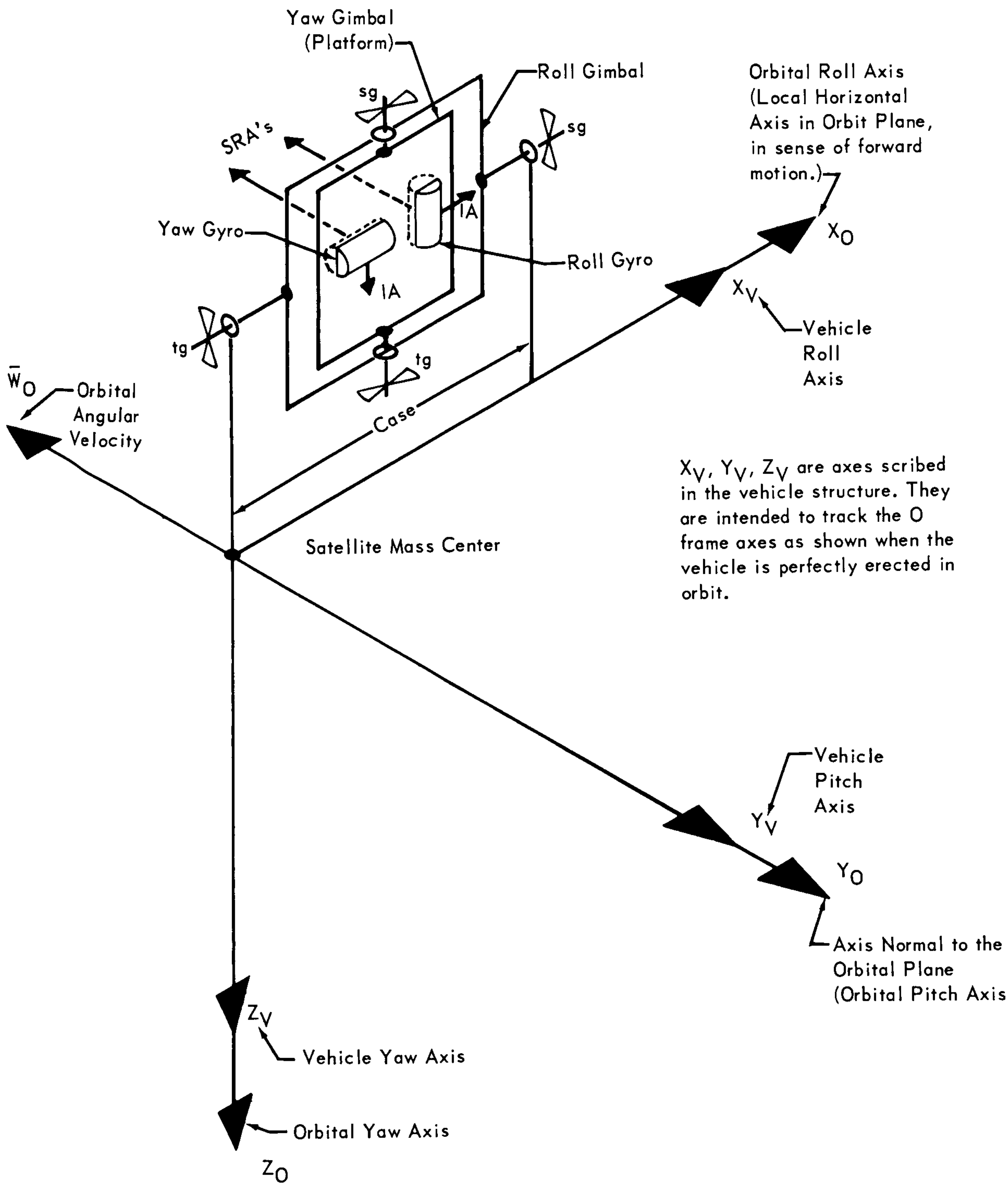


FIGURE 3.3-3

Figure 3.3-4. In general, the previous discussion on experiment procedure and orbital and vehicle control requirements is also applicable to the OVHI. The OVHI is presently being developed under a Sperry sponsored in-house program, and laboratory testing on a completed model is expected to begin in early 1965. The OVHI is expected to weigh 10 pounds and use 25 watts of power.

Test Method - The procedure for conducting the test is outlined below. The test must be performed on an earth-oriented vehicle which is also controlled in yaw approximately to the orbit plane. The gyrocompass is off through launch and until the vehicle is stabilized in orbit to local vertical/orbit plane reference. Following stabilization, the gyrocompass is turned on. The vehicle is controlled so that, as near as practical, the attitude errors sensed by the horizon sensor are nulled. Gross control ($\pm$ 10 degrees) of vehicle yaw to the orbit plane is also required to permit orbit plane acquisition by the gyrocompass. A roll attitude error from the horizon sensor is supplied to the gyrocompass electronics to start the gyrocompassing operation. The difference between the horizon sensor and platform roll error signals is used to torque both the roll and yaw gyros and, hence, the platform gimbals for correct alignment.

One week is considered an adequate duration for the total test. Initially, the gyrocompass and instrumentation remain energized for one day. Operational parameter data such as temperature are collected continuously throughout each orbit. Gyrocompassing accuracy data is collected only during the day portion of each orbit when the sunlight enters the master reference sensor field-of-view. After one day, the equipment is turned off and remains off for 2 days. The equipment is then turned on and the gyrocompassing test is continued for four days. The purpose of this off-on test is to demonstrate correct gyrocompassing operation after an appreciable off time in orbit. During the final four days of

OVHI SCHEMATIC

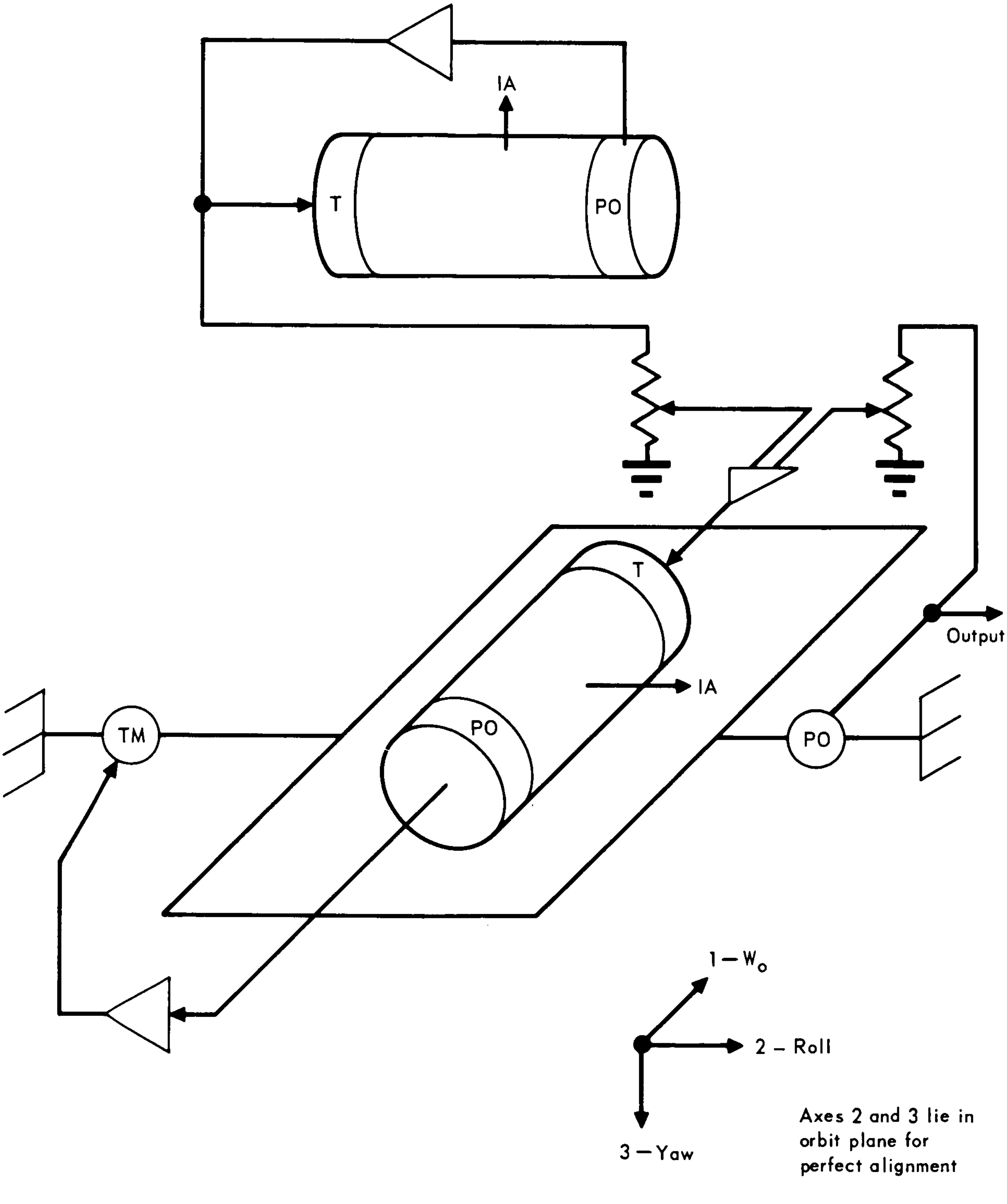


FIGURE 3.3–4

operation, in addition to determining gyrocompassing accuracy, enough data is collected to permit estimates on gyro drift rates.

Major Error Sources - The two primary categories of error sources affecting gyrocompassing accuracy and its determination are: (1) systems errors, and (2) experimental errors. System errors include horizon sensor accuracy, gyro drift, spacecraft stabilization and orbit ellipticity. Experimental errors are readout errors, master reference uncertainties, and orbit determination from ground tracking.

Horizon sensor measurement errors have approximately a one to one correspondence with gyrocompass errors depending upon loop parameters. This error is expected to be about 0.1°. Gyro drift can contribute about 0.25° error after one hour operation. This assumes a drift rate of 1°/hour and an orbit rate of 240°/hour. Advances in the state of the art in gyro technology should minimize this source of error in the near future. The spacecraft is required to be controlled to within $\pm 5^{\circ}$ of the local vertical. Since the gyrocompass platform is fixed relative to the vehicle, orientation errors will affect the gyrocompass. This error is small and will not exceed 0.1°.

The experiment errors are smaller than the system errors. Readout error should not exceed 0.1°. Master reference uncertainties and orbit determination errors are small compared to the 0.1 readout error.

Assuming the use of precision devices, the horizon sensor accuracy, primarily a function of horizon anomalies, would seem to be the limiting factor in this experiment.

3.3.4 Experiment Physical Parameters - Dimensions, weights and power requirements of experiment equipment are given in Table 3.3-2.

TABLE 3.3-2

GYROCOMPASS EXPERIMENT PHYSICAL PARAMETERS

SYSTEM COMPONENT	SIZE		WEIGHT (LB.)	POWER	
	VOLUME ($FT.^3$)	L – W – H (IN.)		PEAK (WATTS)	NOMINAL (WATTS)
Experimental Device Gyro Compass and Experiment Electronics Pkg.	.296	8 x 8 x 8	21	30	20
Support Equipment Signal Conditioner Power Supply	 .00463 .0625	 2 x 2 x 2 3 x 6 x 6	 .25 5	 1 44	 1 30
Total	0.363		26.25	44	30

3.3.5 Data Parameters - Table 3.3-3 lists the important experimental parameters to be measured during the test. All signals are analog with a sample rate of one sample per second. The accuracy required of the experiment equipment is 1% except for the master reference (sun or star tracker) for which 60 arc seconds is called out. Vehicle rates need only be measured to within 2.5%. All signals listed are essential, continuous and should be recorded.

3.3.6 Vehicle Orbit and Attitude Requirements - The test, as presently conceived, is insensitive to orbit inclination. Gyrocompassing accuracy will deteriorate for extremely high altitudes and for eccentricities much greater than 0.1. If the orbit is selected purely on the basis of this experiment, a near circular orbit from 150 to 300 n.m. is preferred.

The vehicle must be controlled to an earth-orbit plane orientation. Permissible attitude deviations are in the 2 to 5 degree range with tighter control

TABLE 3.3-3
GYROCOMPASS EXPERIMENT DATA PARAMETERS

DATA POINT	SIGNAL FORMAT	PARAMETER RANGE	SAMPLE RATE (BANDWIDTH)	ACCURACY	RECORD	ESSEN-TIAL	CON-TINUOUS
Gyro Compass							
Yaw Gimbal Position	Analog	± 10 degrees	1 sample/sec.	1%	X	X	X
Roll Gimbal Position		± 10 degrees			X	X	X
Yaw Gyro Temp.		± 10°F			X	X	X
Roll Gyro Temp.		± 10°F			X	X	X
Electronics Temp.		± 50°F			X	X	X
Power in		off/on		-			
Sun Tracker Angles		70°		60 sec.	X	X	X
Vehicle Rates		1°/sec.		2.5%	X	X	X
Vehicle Attitudes				1.0%			
Yaw		± 15°			X	X	X
Pitch		± 1°			X	X	X
Roll		± 5°			X	X	X

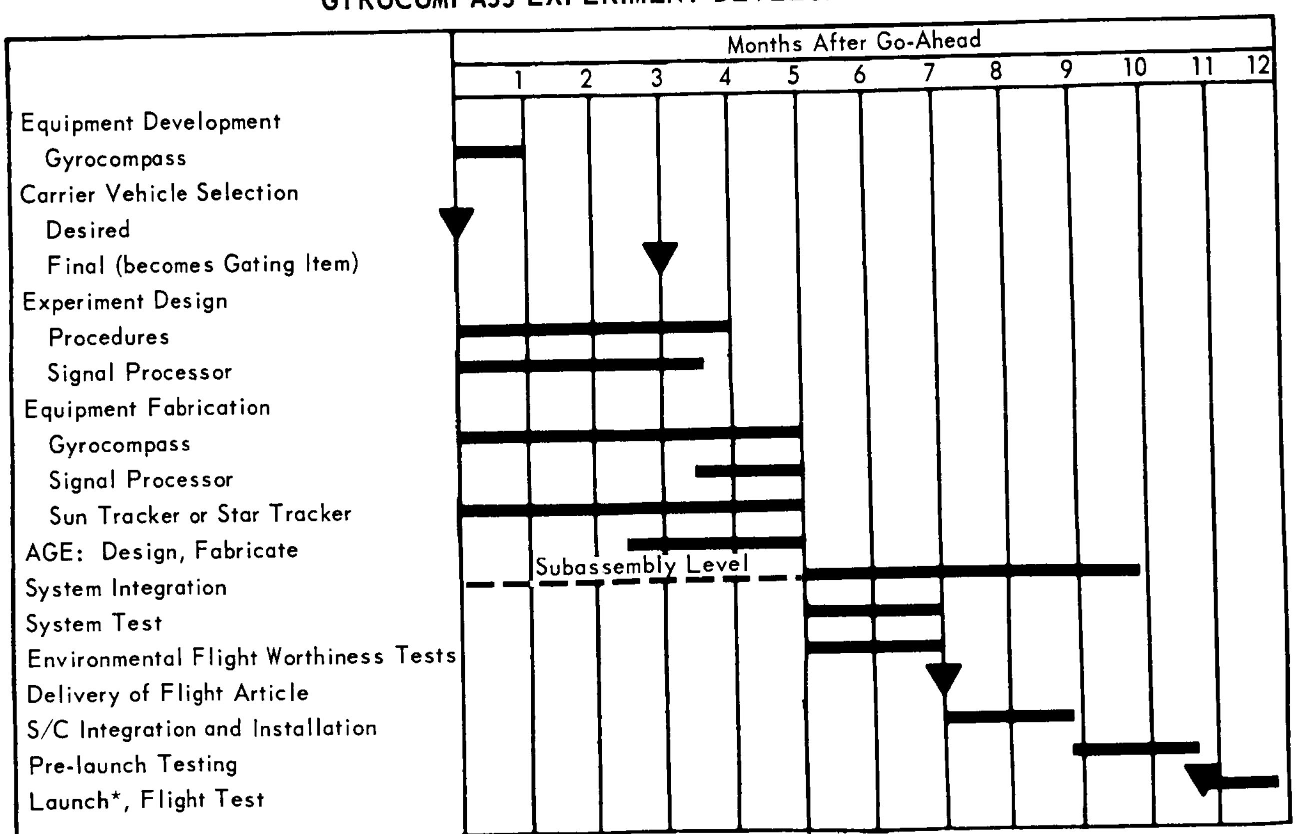
(2 degrees or less) required in vehicle roll. Attitude rates should be held to less than 0.1 degrees/second since the gyrocompassing accuracy goal is 0.1 degree and the data sample rate will be in the range of one sample per second. Furthermore, even though the gimballed gyrocompass is intended to filter the effects of body rates relative to an earth-orbit plane coordinate system, there are time constants associated with these filtering loops which might affect the overall readout accuracy in the presence of high body rates.

3.3.7 Experiment Support and Data Handling Requirements - Either a sun tracker or star tracker master reference will be used to evaluate gyrocompass accuracy. The details of both are discussed in Appendix A. The gyrocompass platform is mounted to the vehicle and oriented with respect to a set of vehicle axes. It has limited freedom of motion in roll and yaw. 3600 watt hours are required to operate the experiments equipment during the five days of continuous operation. Only two ground commands will be required, the first to initiate the test, the second to dump the data. Data handling for the experiment is discussed in Appendix B.

3.3.8 Experiment Flight Test Plan - The experiment development plan is shown in Figure 3.3-5. Delivery of the flight article can be made after 7 months assuming five months for equipment fabrication and two months of system and environmental testing. Assuming carrier vehicle availability and no extended delays, the entire test could be completed 12 months after go-ahead.

The orbital test procedure is quite simple and will not require a programmer. The flight test sequence shown in Figure 3.3-6 includes the previously mentioned ground commands to initiate the test and dump data. Test initiation involves

GYROCOMPASS EXPERIMENT DEVELOPMENT PLAN



*Depends on Carrier Vehicle – may vary from 1 month to 8 months after installation.

FIGURE 3.3-5

GYROCOMPASS EXPERIMENT FLIGHT TEST SEQUENCE

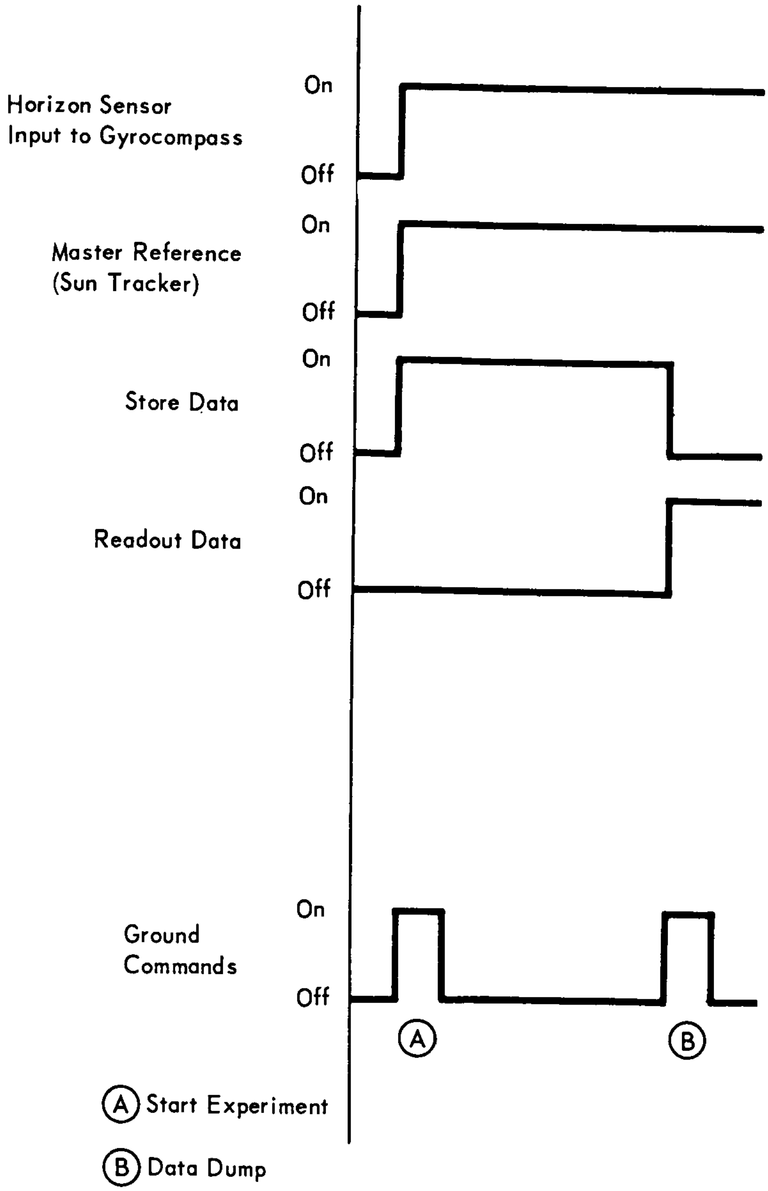


FIGURE 3.3-6

closing the loop between the horizon sensor and gyrocompass, activating the master reference, and storing data.

REFERENCES

1. NASA TND-1771, "Some Control Problems Associated with Earth-Oriented Satellites" by Vernon K. Merrick, June 1963.

2. AIAA Paper 64-238, "An Orbital Gyrocompass Heading Reference for Satellite Vehicles" by Robert L. Gordan, June 1964.

3. NASA TND-2134, "The Use of a Two-degrees-of-freedom Gyroscope as a Satellite Yaw Sensor" by Francis J. Moran, February 1964.

4. Aviation Week and Space Technology, 10 July 1961, "Nimbus Uses Wheels, Jets for Control", by George Alexander.

5. Sperry Gyroscope Company Experiment, "Orbital Vehicle Heading Indicator", 9 November 1964.

6. MIT/IL R-422 (confidential), Interim Technical Documentary Report for Contract AF04(695)-289, 10 September 1963.

3.4 Orbital Testing of High Reliability Horizon Sensors

3.4.1 Objective - The objective of this experiment is to measure the in-flight performance characteristics of a high reliability type horizon sensor. The experiment is particularly applicable to those horizon sensors with an accuracy requirement approaching 1° (for testing of precision horizon sensors, see the Horizon Sensor Accuracy Experiment, Paragraph 2.5).

3.4.2 Background - In recent years a large number of proposals on high reliability, earth local vertical horizon sensors have been submitted within the aerospace industry. These sensors were proposed as prime reference systems for large deadband attitude control loops, for antenna pointing reference systems and many other uses which require a local vertical reference in space vehicles. The high reliability units used have generally been linear scan (edge tracking) such as used on the Orbiting Geographical Observatory or illumination comparison (radiometric balance) such as used on Ranger. Many other types, with and without moving parts, have been studied and proposed. In general it has been concluded that the performance of units used has been satisfactory, but measurements of the in-flight performance characteristics as referenced to a precision system are not available. More important to the industry perhaps is the fact that many new ideas are not being used, since the available "tried and true" systems are a "known" quantity. Several vendors have proposed major advances in the design art, only to be turned down because of the unknown or unproven capability of the proposed equipment.

The type of experiment proposed here is ideally suited to testing in orbit, against a real earth target. This would permit an evaluation of an advanced state-of-the-art concept before being selected as a prime reference system for a spacecraft. The Horizon Sensor Accuracy Experiment, Paragraph 2.5, requires the use of a high precision star tracker as a Master Attitude Reference system. The complexity of data handling, data reduction and programming does not lend itself readily to repeated, multiple tests of design concepts. For this experiment, a precision horizon sensor serves as the master attitude reference; in fact, the carrier vehicle prime reference system may be satisfactory for determination of the test sensor performance characteristics.

A detailed evaluation of sensors is required in order to determine:

a. Accuracy and repeatability

b. Susceptibility to disturbances

c. Output variation due to earth atmospheric anomalies

d. Life characteristics in an orbital environment.

Accuracy characteristics and effect of earth anomalies can only be tested in orbit. Lack of detailed knowledge of the earth as an infrared (IR) target has made it virtually impossible to build a simulator which realistically takes into account all the various error producing anomalies. Each sensor design is susceptible to some extent to each of the various types of anomalies. Simulator design parameters which must be taken into account are temperature, gradient slope, cold-cloud effects, seasonal variations, day-night variations and the effect of the orbital environment. Each of the parameters is discussed in detail in Paragraph 2.4.2. For the no-moving part sensor, the exposure to a near zero-g environment is relatively unimportant. The effects caused by solar radiation and other factors could be simulated, but simultaneous simulation of all the parameters is difficult and costly.

In summary, if sufficient knowledge were available on the earth's infrared gradient characteristics, a test facility could be designed to test each type of horizon sensor. The facility would be large, expensive, and difficult to operate. Some of the variables discussed above may be relatively insignificant to the performance of the unit, depending upon the design. However, one should keep in mind that even small factors, when accumulative, can produce large variations. A flight test of the unit in question would provide, at a reasonable cost, an evaluation of the total performance when exposed to all the variables of the orbital conditions and the infrared characteristic of the actual earth.

3.4.3 Functional Description - Determination of the accuracy of a horizon sensor requires a master attitude reference system, a stabilized earth referenced vehicle, a recorder for data storage and a test unit. A functional block diagram of the experimental package for measurement of a horizon sensor's performance is shown in Figure 3.4-1. After orbital insertion, the test horizon sensor outputs, vehicle attitude and rate, vehicle horizon sensor outputs and other selected parameters are recorded for data retrieval over tracking station sites. The recorder is programmed to record either continuously or in sequenced on-off time cycles to provide a large statistical sample of the horizon sensor performance. Information on sensor performance with seasonal variations, day-night variations and systems operational parameter variations over a wide range of longitudes and latitudes is obtained. System operational parameters are either closely maintained or thoroughly monitored in order that the effect of variables may be carefully evaluated.

Through use of ground command signals, either the recorded data may be transmitted in compressed or real time. Due to the limited frequency response characteristics of space qualified recorders, only a few carefully selected parameters are recorded. Additional test points and parameters will be available during real-time transmission. This technique is utilized to assure that information obtained from the experiment will establish the accuracy of the sensor and also will provide data on which to establish the design changes necessary to improve the horizon sensor design.

EXPERIMENT BLOCK DIAGRAM

HIGH RELIABILITY HORIZON SENSOR EXPERIMENT

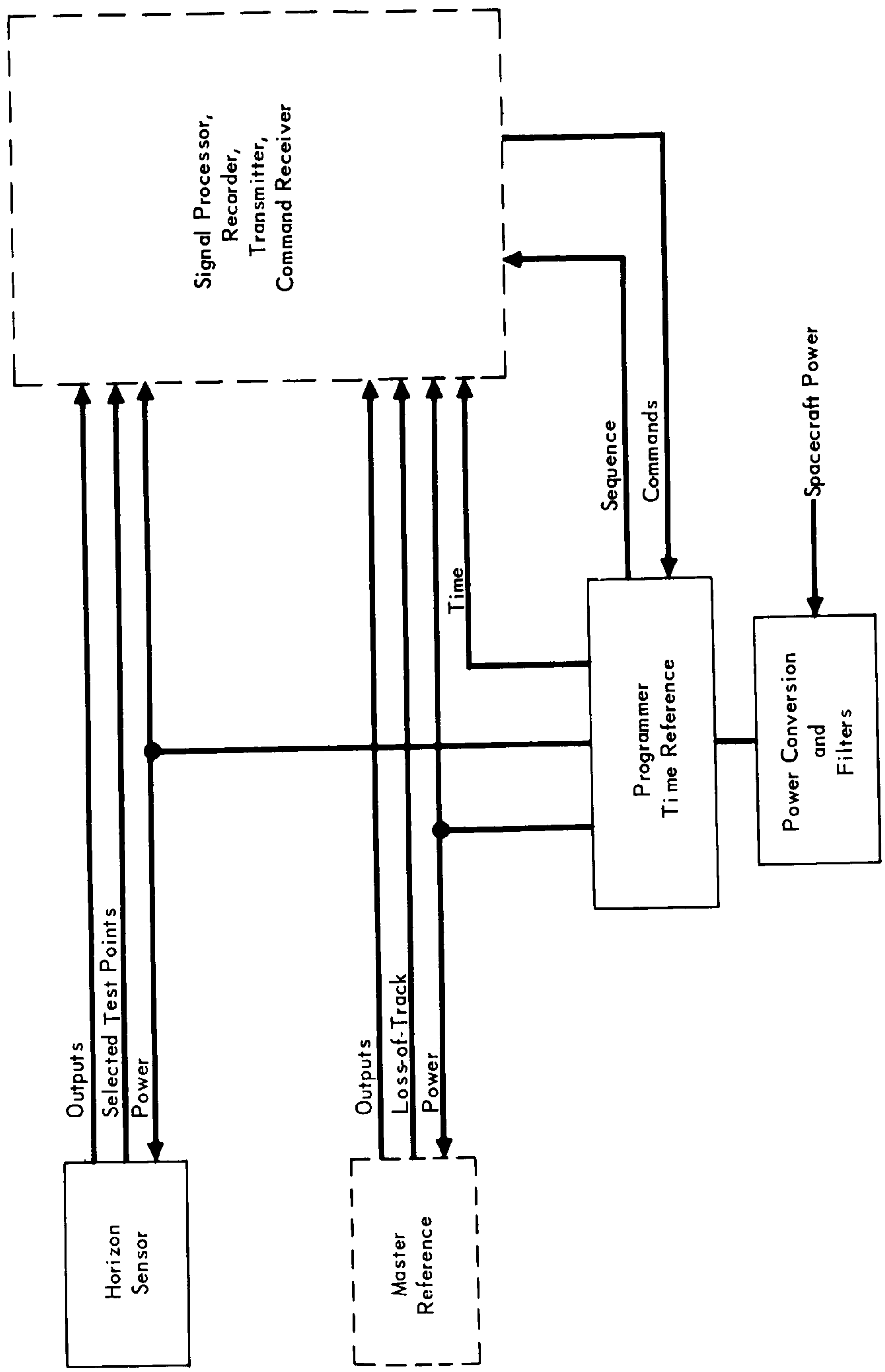


FIGURE 3.4-1

Test Horizon Sensor - The unit to be tested is not critical. Many types are presently available and many others have been proposed. The experiment will be especially useful in evaluating new design concepts or units which have given indications of failure to meet specification requirements, but the cause of failure could not be readily established. For the purposes of this experiment, and in order to provide a rough estimate of the physical requirements of the experiment, the physical parameter and instrumentation requirement tables contain an analysis of a unit proposed by Quantic Industries and the possible test points to be monitored.

Attitude Reference Horizon Sensor - In order to obtain a detailed evaluation of the performance characteristics of the low accuracy horizon sensor in orbit, an accurate determination of the vehicle attitude is required. The reference system should provide, ideally, a reference or order of magnitude better than the required performance of the unit under test. Since this experiment is primarily intended for the evaluation of sensors with an accuracy of 1° (or greater) at null, the attitude reference should be accurate to 0.1°; however, an accuracy of 0.3° only introduces 4 percent error in the analysis.

Since an accuracy of 0.1° to 0.3° will suffice for this experiment, two approaches may be taken. First, a spacecraft with an on-board precision horizon sensor could be utilized as the test vehicle. The on-board horizon sensor-IMU (platform or strap down) system will be used to establish an accurate measure of the spacecraft attitude with respect to local vertical. Secondly, for those spacecraft with low precision master attitude reference, a high precision horizon sensor is added to the experiment package to act as the reference. In either event, the test horizon sensor and the precision horizon sensor outputs are compared and analyzed to evaluate the test units performance characteristics. Further discussion of Master Attitude Reference Systems is contained in Appendix A.

If the spacecraft does not already contain a precision attitude reference system, a sensor must be added to provide the required instrumentation. A modified Gemini Horizon Sensor, an OGO type horizon sensor, a Saturn type horizon sensor or an Agena type horizon sensor could be utilized, depending upon the altitude, ellipticity and attitude range of the vehicle to be used for the experiment. The sensor should be designed or modified to operate in the 14-16 micron spectral pass band in order to reduce the effects of anomalies to a minimum. Sensor pitch and roll outputs, loss-of-track (or ignore signal), input power, temperature and selected test points are monitored and telemetered to assure proper operation.

Test Method - After orbital insertion and vehicle stabilization, the experiment package is turned on. Due to the limitation of data which a practical size recorder can accumulate, the recorder is operated in on-off cycles. During those orbits when the vehicle passes over a tracking station the recorded data is dumped or retrieved. After taking data for three to five orbits the experiment is turned off. After an off period of three to five days the experiment sequence is repeated. The process of on for three to five orbits and off for three to five days is continued for a test period of no less than one month. If a high confidence in the reliability of the unit under test is desired, the test units power should be left on throughout the entire orbit time and, in addition, the performance evaluation should continue for a minimum of one mean life expectancy for the unit under test.

3.4.4 Experiment Physical Parameters - The physical and operating parameters of horizon sensors which have been and could be considered for this experiment cover a wide range. Tabulation of only a representative case is included herein in view of the objective to define a test which would evaluate new as well as existing designs. Table 3.4-1 contains the tabulation of a representative experimental package. The table includes the parameters of a master reference system to be used in those cases where the carrier vehicle does not have an on-board precision horizon sensor.

3.4.5 Data Parameters - Table 3.4-2 is a tabulation of the signal parameters to be telemetered for this experiment. The priority or relative need for the signal is indicated by the notation essential or desirable in the remarks column. Additional remarks indicate whether the signal is to be recorded or available for real-time transmission and whether or not the signal requires continuous monitoring to obtain short term or transient information.

The information required from this experiment is a measure of the performance characteristics of a horizon sensor. To obtain this data requires a precision reference system. The signal parameters on vehicle attitude, to be measured through use of a master reference system or the on-board attitude reference system, are also listed in Table 3.4-2.

3.4.6 Vehicle Orbit and Attitude Requirements - The requirements on vehicle attitude control and orbit parameters are essentially identical to the requirements of the Horizon Sensor Accuracy Experiment defined in Paragraph 2.5. Table 2.5-4 contains an analysis of the effect of parameter variation.

3.4.7 Experiment Support and Data Handling Requirements - Performance of this experiment requires interfacing with the carrier vehicle equipments. Signals from the experiment must be conditioned for transmission; a master reference must be provided; data must be recorded; ground command signals must be processed; and power must be made available. A discussion of master reference techniques is contained

TABLE 3.4-1
HIGH RELIABILITY HORIZON SENSOR
EXPERIMENT PHYSICAL PARAMETERS

ITEM	VOLUME CU.FT.	VOLUME L – W – H (INCHES)	WEIGHT (POUNDS)	AVERAGE POWER (WATTS) PEAK	AVERAGE POWER (WATTS) NOMINAL
Experiment Sensor	.15	5 x 5 x 10	7	12	10
Support Equipment					
1. Program Sequence Electronics	.03	5 x 3 x 3	4	10	10
2. Signal Conditioning	.03	5 x 3 x 3	4	5	5
3. Power Supply and Filters	.05	5 x 5 x 3	5	5	5
4. Recorder	.3	14 x 9 x 5	15	40	40
Subtotal	.56		35	72	70
Master Reference* Horizon Sensor	.2		12	12	12
TOTAL	.76		47	84	82

*Required if not on carrier vehicle.

in Appendix A. Appendix B contains a general discussion of data handling techniques which are applicable to this and other experiments. The magnitude of the interfacing problem is best indicated by the scope of the equipment support areas discussed in the following paragraphs.

Programmer and Time Reference System - This unit is a "special" design to serve the purposes required by this experiment. It will consist of the necessary electronics, relays, time reference system and logic networks to:

a. Update time reference system upon ground commands,

b. Sequence data transmission to ground stations at predetermined times or upon ground command.

c. Turn systems on and off either upon ground command or in a preprogrammed time sequence.

TABLE 3.4–2

HIGH RELIABILITY HORIZON SENSOR EXPERIMENT DATA PARAMETERS

DATA POINT	SIGNAL FORMAT	RANGE OF PARAMETER	SAMPLE RATE (FREQ. RESP.)	ACCURACY	RECORD	ESSEN-TIAL	DESIR-ABLE	CON-TINUOUS
1. Horizon Sensor (Test)								
Pitch Output	Analog		3 CPS	± 1%	X	X		X
Roll Output	Analog		3 CPS	± 1%	X	X		X
Loss-of-Track (Ignore)	Digital	On/Off			X	X		X
Temperature	Analog	0–160°F		± 5%	X		X	
Input Power	Analog	28 VDC		± 5%	X		X	
Internal Power Supplies	Analog			± 5%	X		X	
Pre-Amp Output	Analog			± 5%	X		X	
2. Reference Horizon Sensor								
Pitch Output	Analog		3 CPS	± 1%	X	X		X
Roll Output	Analog		3 CPS	± 1%	X	X		X
Loss-of-Track (Ignore)	Digital				X		X	
Input Power	Analog	28 VDC		± 5%	X		X	
3. Spacecraft Signals								
Attitude Gyro Signals (3)	Analog		3 CPS	± 1%	X		X	X
Rate Gyro Signals (3)	Analog		2 CPS	± 1%	X		X	X
Spacecraft Power	Analog			± 1%	X		X	
Time Reference	Digital			± 1 Bit	X	X	X	

d. Provide a precision time reference,

e. Sequence recorder on and off in a preprogrammed sequence, and

f. Sense failures of experimental package which would affect performance of the primary vehicle and power-down the system as required.

Power Conversion and Filters - The power conversion and filter package provides the necessary RFI filtering and power conversion (DC to AC, DC to DC and voltage regulation) required by the individual experiment systems. The size of this system will be a function of the capability of each system vendor to provide the necessary power conversion and filtering in each system within the confines of the existing hardware packages and the allowable time scale to provide compatibility between the experiment and spacecraft power.

Recorder - In order to obtain the maximum amount of data, and in order to obtain data from areas where ground tracking stations are not available, an on-board recorder is required. The selected data will be stored at a low speed and, upon ground command, will be dumped at a high speed to the ground tracking station. The maximum number of channels feasible should be taped in order that data reduction and performance evaluation is as unhampered by a lack of knowledge of parameter variation as possible.

Signal Conditioning - For those signals not already modified to be compatible with the telemetry system, signal conditioning will be required. Only a limited number of channels are recorded and the bandpass frequency requirements are not overly stringent. The signals will be commutated and encoded for recording and transmission.

Environmental Control - One of the purposes of this experiment is to evaluate new design concepts in horizon sensors. In some cases it may be desired to test engineering prototype units which will require a relatively close control on temperatures. Coldplates and/or heater elements may be required; however, a thorough analysis of the thermal characteristics of the carrier vehicle will be required to determine the needs.

Mounting - It will be necessary to mount the test horizon sensor (and the reference horizon sensor where used) in such a manner that an unobstructed field of view is available. Precision alignment of the horizon sensor with respect to the master reference is required.

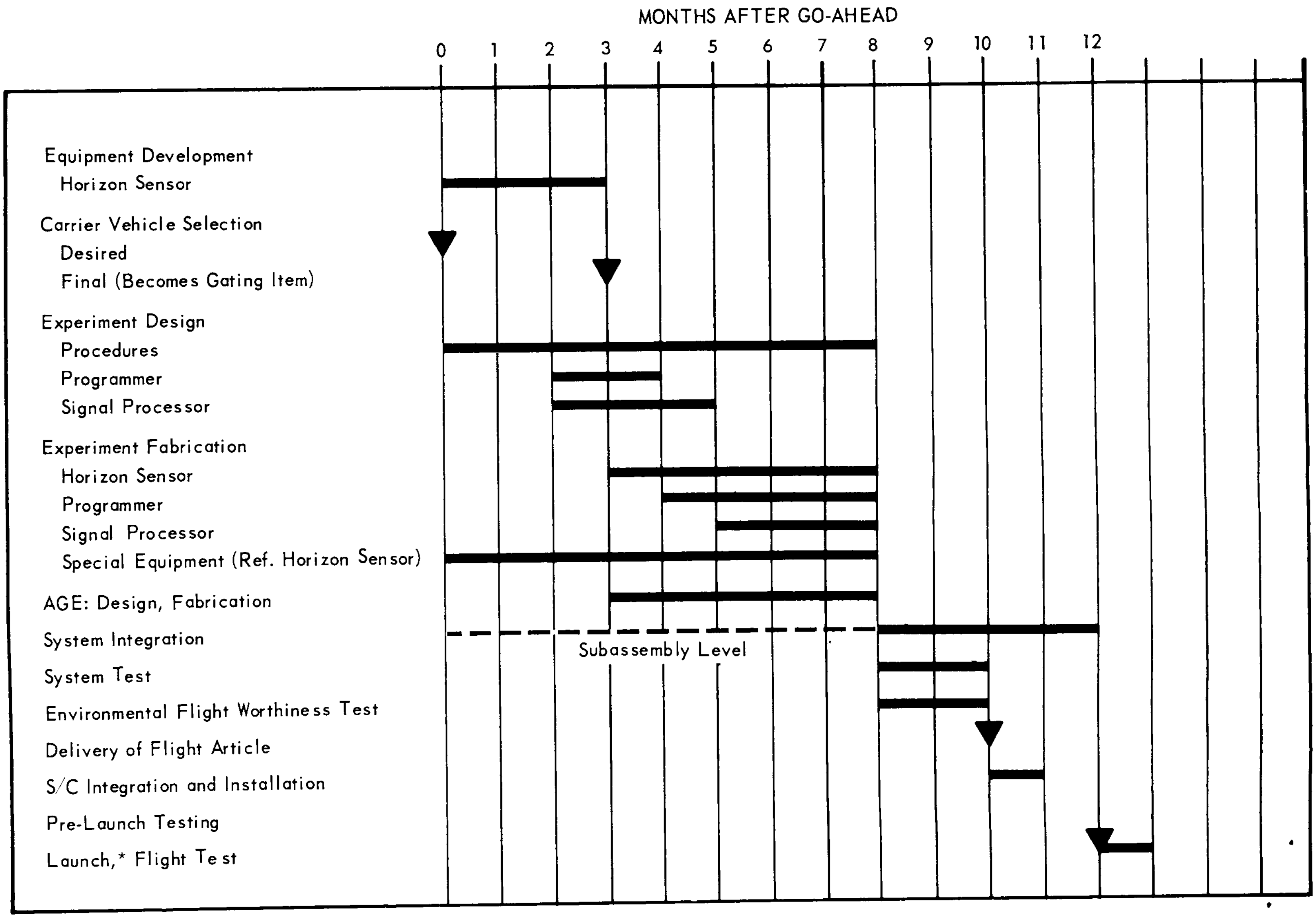
Electrical Energy Requirements - The estimated energy for the proposed two week accuracy evaluation portion of this experiment is 3,040 watt-hours. The master reference, if required, uses an additional 800 watt-hours of energy in the same time span. Life testing requires 3,040 watt-hours per week (79,000 watt-hours for six months) for the horizon sensor.

Ground Command Requirements - Ground commands are utilized to turn the experiment on or off, command data retrieval, update or revise the operating sequences and command real-like transmission of the system analysis test points. It is not expected that these four commands will cause unusual loads on the ground station capabilities.

3.4.8 Experiment Flight Test Plan - Figure 3.4-2 is a flow diagram of the experiment development plan. The schedule is based on the required development, integration and flight time and is intended for planning purposes. The time for development of the horizon sensor is an estimate and will vary depending upon the design concept and past experience of the manufacturer with similar designs.

The entire schedule is predicted on an early selection of a carrier vehicle. Delays in the selection of the vehicle may affect the schedule inasmuch as thermal and mechanical designs are dependent upon carrier vehicle parameters.

HIGH RELIABILITY HORIZON SENSOR EXPERIMENT DEVELOPMENT PLAN



*Depends on carrier vehicle – may vary from 1 month to 8 months after installation.

FIGURE 3.4–2

Flight Test Sequence - Figure 3.4-3 is a time plot of the operating sequence for three orbits. Figure 3.4-4 illustrates the mission cycles for a two-week mission. It is not necessary to operate the experiment continuously while in orbit, nor is it necessary to repeat the operating sequence more than once. The sequence presented is for obtaining a relatively large amount of statistically distributed data over a two-week period. The suggested sequence is as follows:

a. After orbital insertion and vehicle stabilization the experiment is turned on.

b. The recorder is operated in cycles of twenty minutes on, twenty minutes off for three orbits.

c. The experiment is placed in a standby or off condition for three days.

d. Steps (a) through (c) are repeated.

e. Steps (c) and (d) are repeated three times.

This sequence will provide 540 minutes of test data which covers a wide range of latitudes and longitudes, day-night variations and seasonal variations (polar orbit only). Six months of operation in a near equatorial orbit will provide data on target variations as well as life characteristics of the sensor.

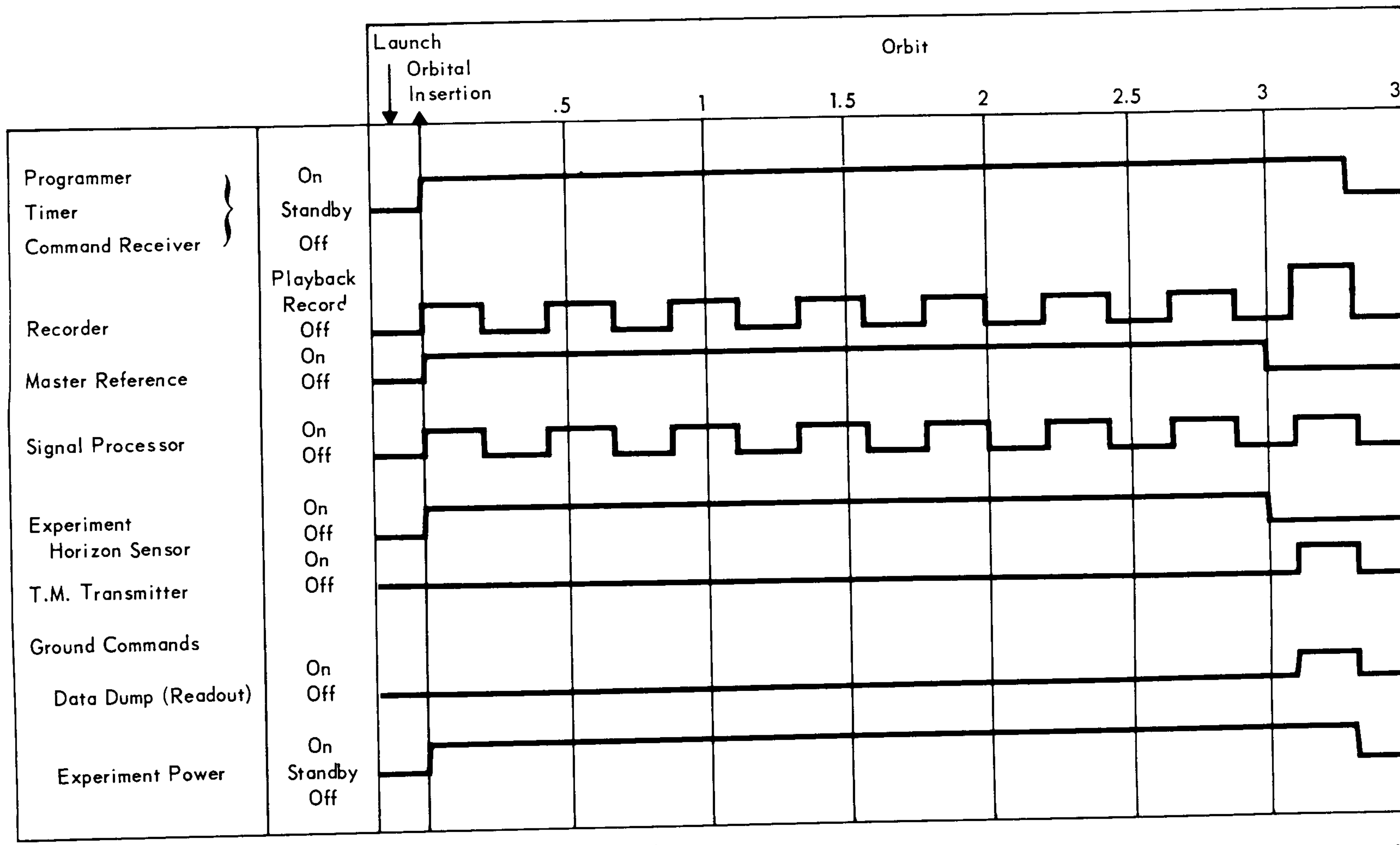
Post Experiment Data Reduction - The data reduction sequence is as follows:

a. Determine, through comparison of the test horizon outputs and the master reference system outputs, the test horizon sensor error.

b. Process the error data to obtain a statistical analysis of the horizon sensor performance capability.

If the error analysis indicates performance capabilities as good or better than the reference system, an experiment similar to Experiment 2.5 should be performed.

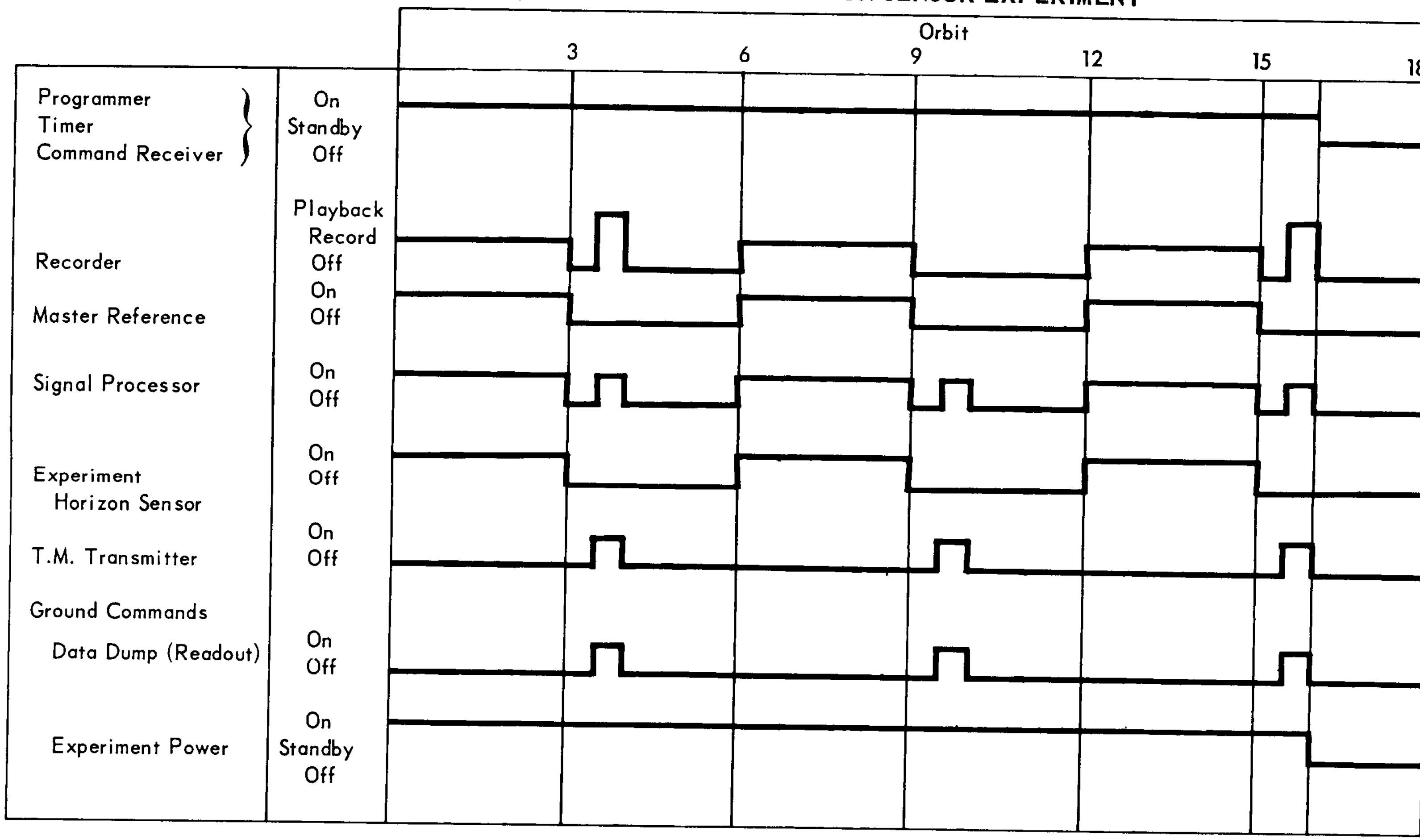
FLIGHT TEST SEQUENCE, HIGH RELIABILITY HORIZON SENSOR EXPERIMENT



Note: See Figure 3.4–4 for mission cycles.

FIGURE 3.4–3

MISSION CYCLES, HIGH RELIABILITY HORIZON SENSOR EXPERIMENT



NOTE: Repeat cycle during orbits 48 through 64, 96 through 112 and 144 through 160.

FIGURE 3.4-4

3.5 Title - Star Recognition

3.5.1 Objective - The purpose of this experiment is to evaluate the in-flight capability of a star-mapper system to automatically provide vehicle attitude information.

3.5.2 Background - Recognition of stars by man has been accomplished for centuries by pattern or constellation recognition and star brightness (magnitude). Once particular stars have been recognized, local position can be precisely determined through use of stellar navigation techniques.

To determine an orbiting vehicle attitude or position in space, stellar navigation can be used. With automatic equipment and without "man-in-the-loop", recognition becomes a problem. Relative magnitude can be ascertained through a comparison technique utilizing the sun (after filtering and attenuation) as a reference (Reference 1) or by using a lamp giving known illumination as a reference. Logic circuits utilizing automatic gain control (AGC) voltages in the tracking loop can also be used to some extent for magnitude recognition. Color evaluation can be accomplished by utilizing spectrally sensitive detectors or by usage of multicolor filter techniques as discussed in Paragraph 2.7 (References 2 and 3). Pattern recognition requires further refinement however, and is not quite as straight forward.

If the vehicle is earth-oriented with a known stability and its approximate position is known, then the approximate line-of-sight to a given star is known. If a star tracker is pointed to the line-of-sight and is commanded to search an angle which is about twice the position and attitude uncertainty, the required star will be acquired.

Another technique, which is to be used on the OAO, Surveyor and Mariner vehicles is to acquire the sun, align the vehicle roll axis to the sun, and while maintaining the pitch and yaw axes controlled to the sun line, roll the vehicle. On OAO, 4 to 6 star trackers are slewed to predetermined positions, and when all star trackers simultaneously acquire, lock-on is commanded and roll is stopped. On Mariner and Surveyor, the vehicle is rolled about the sun line; however, lock-on occurs on only one star, Canopus. Whereas OAO depends upon the unique relationships among the angular displacements of 4 to 6 stars, the Canopus star tracker on Surveyor depends upon relative magnitude. Canopus, the second brightest star as seen from earth, differs by nearly one magnitude from the brightest and the third brightest stars.

Another approach would be to use a wide field of view star mapper which would detect and measure the angle to several stars essentially simultaneously. With the measured angular separations of all stars stored in the computer, a search for the viewed star pattern would be initiated. Once the pattern is recognized, the computer would present an output signal which is a function of the spacecraft position and attitude with respect to the stars (or earth, with additional computation).

Star mappers and star trackers have been utilized for a number of years on aircraft, missiles and shipboard installations. The units provide the necessary reference system for autonomous navigation, but there is usually a man-in-the-loop for decision making (with missiles it is immediately prior to launch). Long term unmanned orbital flights have a different problem insofar as decision making and star recognition is concerned; if a star tracker is used, a computer is required for the decision making and recognition. Due to the limited accuracy of horizon sensors, even some earth oriented vehicles would benefit with the utilization of a precision stellar reference system.

In order to utilize the stars as a reference system, several techniques have been devised. One type is the gimballed, small field of view star tracker. A telescope with a small (usually 1°) field of view, a detector and a servo tracking loop are mounted on 2 or more gimbals. The gimballing provides the necessary "large field-of-view" for pointing the telescope over a large area of the celestial sphere. The unit is designed to track one star at a time and attitude information is derived by measuring the gimbal angles with respect to the vehicle axes. By comparing the gimbal angles to stored coordinates, vehicle attitude can be established. By utilizing two or more star trackers (or by sequentially tracking two or more stars with a single star tracker with the aid of a gyro for short term attitude memory) position information may be derived.

Another technique of obtaining attitude and position is to utilize a star mapper. The star mapper is essentially a device with a telescope having a large field-of-view (usually greater than a 60° cone) with a detector system which senses the position of two or more stars essentially simultaneously. The unit is usually body-mounted. The star mapper may or may not have moving parts, depending upon the particular design of the detector system.

Specific detector, signal processing and tracking loops vary a great deal from manufacturer to manufacturer, but the overall purpose of each is essentially the same - to detect stars of a given color magnitude (Reference 4) and provide an output which is a measure of the vehicle attitude and position with respect to the stars (Reference 5 and 6).

Orbital testing of star trackers and star mappers is required to obtain a final or proof evaluation of performance. The extent of ground testing is limited insofar as realistic simulation of actual star magnitude, spectrum and stellar background light. Due to atmospheric attenuation of the star light, especially in the blue and ultraviolet regions, star mappers and trackers designed for orbital use cannot be fully evaluated from the ground using the actual sky as the target. Simulation of such effects as stellar background light, due to the extremely large number of low light level stars distributed in the milky way, is nearly impossible. The star magnitudes when taken individually are below the threshold of detection, but when taken as a group the combined light flux may be sufficient to produce erroneous tracking and incorrect coordinate information.

The differences between the simulation techniques used and the actual star field as viewed in orbit may be sufficient to produce improper tracking of a star tracker or erroneous signals in the star mapper system. A final evaluation can be obtained by testing the units in orbit.

Star tracker units are to be tested, or have been partially tested in orbit. The Mariner Canopus tracker has been flown. The OAO star trackers (gimballed small field-of-view devices) are to be flown in the near future. Some types of star mapper techniques have been evaluated in short ballistic flights, but an orbital test is needed for the star mapper designed with extra-terrestial flight intended.

This experiment is applicable to any of the multitude of proposed star field mappers. The Nortronics star field mapper (Reference 7) is used in the experimental write-up as a representative type of system. Data from this unit will be applicable, at least in part, to several other types of proposed star mapper units. The prime factor involved in selection of this unit is the development status and the availability date of the hardware after go-ahead.

3.5.3 Functional Description - The experimental package for Phase I of the test consists of a star field mapper, a signal processing and encoding unit, a programmer and time reference unit and a power conversion and filter unit. The star mapper is precisely mounted with respect to the vehicle axes. Computer processing of the telemetered star mapper signals will be accomplished after transmittal to ground based facilities. A functional block diagram of the experiment is shown in Figure 3.5-1. Phase I of the experiment is to determine the ability of the star mapper to detect the stars. Threshold and sensitivity will be evaluated, and computer techniques for attitude determination will be verified. Phase II of the experiment requires the addition of a star tracker system in order that the overall accuracy of the star mapper and data processing may be established.

STAR RECOGNITION EXPERIMENT BLOCK DIAGRAM

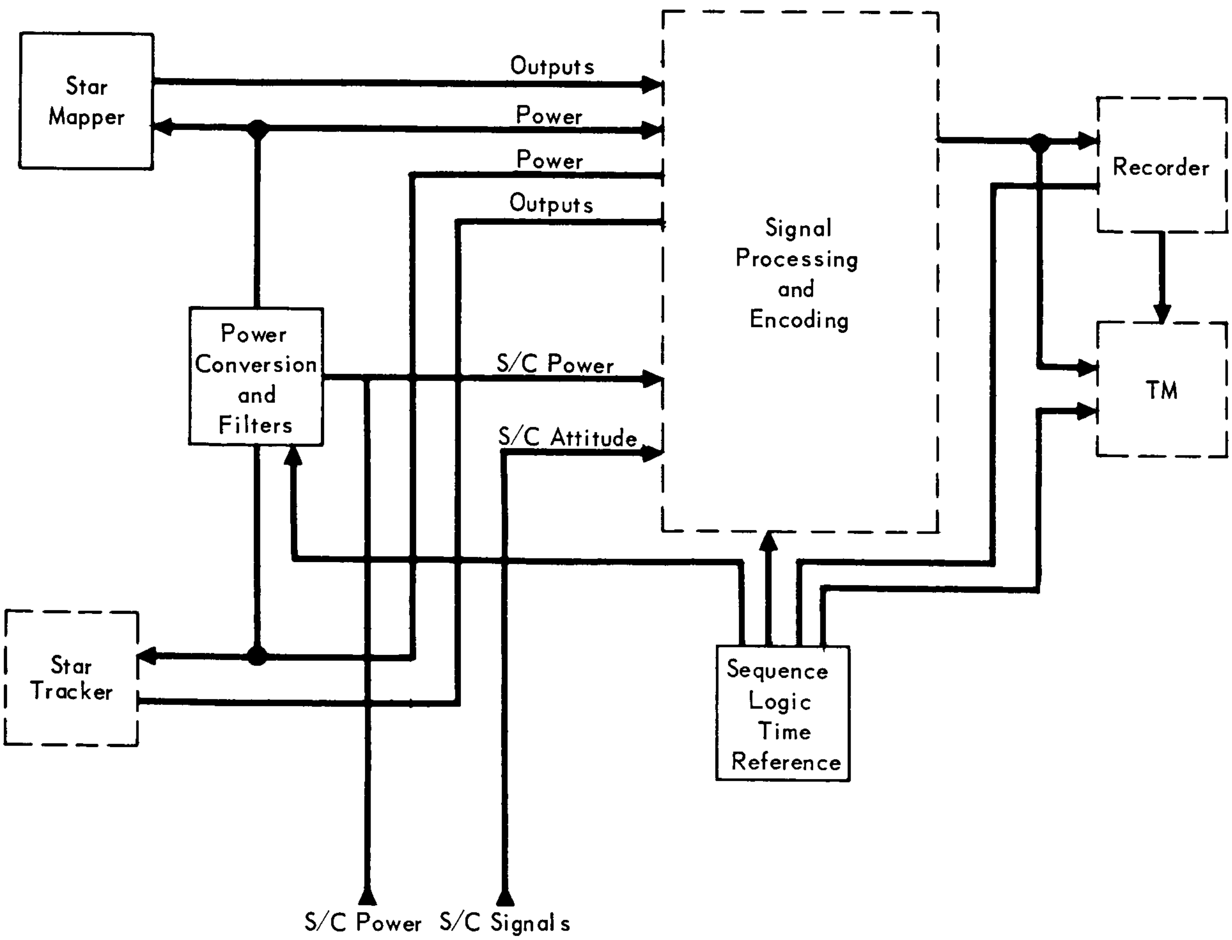


FIGURE 3.5-1

STAR MAPPER FIELD OF VIEW

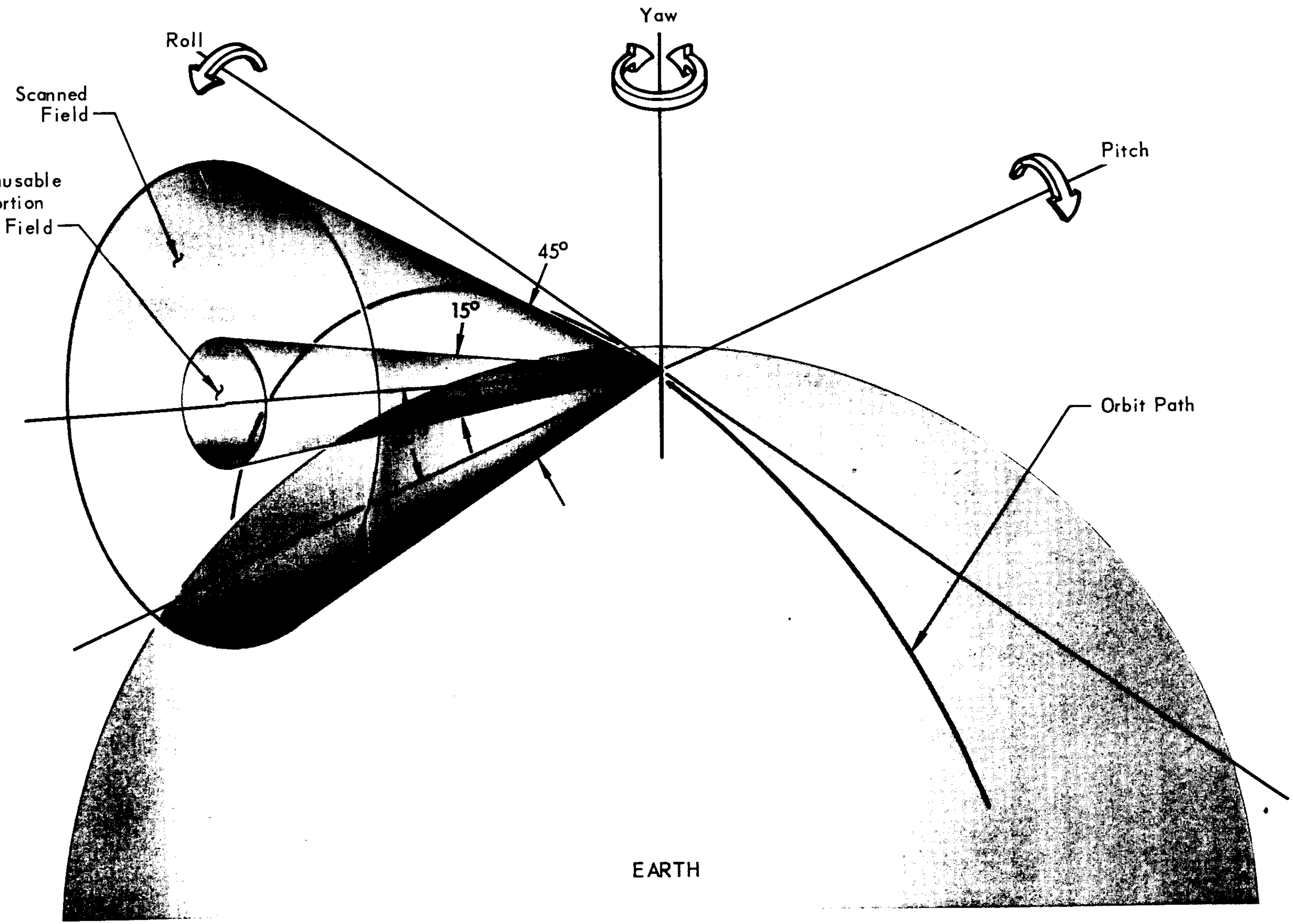


FIGURE 3.5-2

Star Mapper - The Nortronics star field mapper consists of a wide field-of-view optical system, a rotating slit, a motor drive, a photomultiplier detector and signal processing electronics. The optics are designed to cover a 45° conical field of view. The slit obscures a 15° cone in the center of the field-of-view. The unit is mounted such that the center line of the optical axis, hereafter referred to as the line-of-sight (LOS), is out of the orbit plane at an angle of 20° with respect to the pitch axis. Figure 3.5-2 illustrates the field-of-view as viewed along the longitudinal axis of the spacecraft.

The slit-motor drive unit consists of a synchronous motor, precision anti-backlash gear train and a rotating disc with a precision slit. The slit rotates at a rate of 1 revolution per second. A magnetic indicator produces a reference time pulse when the slit passes a reference point during each revolution.

Each time the slit allows star illumination to strike the photomultiplier, an output pulse is obtained. By measuring the time from the reference pulse to the "star" pulse, a measure of the angle to the detected star is obtained. Through use of a computer with stored stellar coordinates and the angles as determined by the star mapper to several stars, the vehicle's attitude may be determined.

Preliminary analysis indicates that a high probability exists that there will be at least two stars of magnitude greater than 2.0 within the field-of-view at all times.

Test Method - After orbital insertion and vehicle stabilization (either inertial or earth oriented) the experiment is initiated. Data is recorded or transmitted periodically depending upon the carrier vehicle mission and orientation. A series of 10 to 20 measurements of 5 to 10 minutes duration each over a period of about 3 days will provide a nominal sample of data for system analysis. Planning of the on-off cycles shall be such that a large portion of the celestial sphere is covered.

Major Error Sources - A detailed error analysis has been performed by Nortronics and only a summary of the results are presented in Table 3.5-1 (Table 5.1 from Reference 7). If the vehicle were inertially stabilized rather than earth oriented, the total RMS error would be 1.18 arc-min normal to the telescope axis and 1.9 arc-min about the telescope axis. The error analysis does not include vehicle rate effects on the error.

TABLE 3.5-1

STAR RECOGNITION EXPERIMENT MAJOR ERROR SOURCES

ERROR SOURCE	NORMAL TO TELESCOPE AXIS (ARC-MINUTES)		ABOUT TELESCOPE AXIS (ARC-MINUTES)	
	RMS	MEAN SQUARED	RMS	MEAN SQUARED
Optical Distortion	.05	.0025	.10	.0100
Slit Eccentricity	.10	.0100	.20	.0400
400-cycle Stability	.25	.0625	1.08	1.1664
Gear Backlash	.25	.0625	.50	.2500
Vehicle Rotation*	.63	.3969	.63	.3969
Background Noise	.60	.3600	1.00	1.0000
Electronic Processing	.15	.0225	.25	.0625
Instrument Calibration	.07	.0049	.75	.5625
Mounting Uncertainty	.50	.2500	.50	.2500
Reference Pulse Uncertainty	.25	.0625	.50	.2500
Sum Mean Square Errors		1.79		3.99
Total RMS Error		1.34 arc-min.		2.00 arc-min.

*Negligible if corrected

TABLE 3.5-2

STAR RECOGNITION EXPERIMENT PHYSICAL PARAMETERS

ITEM	VOLUME CU.FT	VOLUME L - W - H (INCHES)	WEIGHT (POUNDS)	AVERAGE POWER (WATTS) PEAK	AVERAGE POWER (WATTS) NOMINAL
Experiment Sensor Nortronics Star Mapper	.41	13 x 8 x 7	15	18	14.5
Support Equipment 1. Programmer 2. Signal Conditioning 3. Power Supply 4. Recorder	.1 .3	10 x 9 x 2 10 x 9 x 5	5 15	5 40	5 40
Subtotal	.81		35	63	59.5
Master Reference Star Tracker (OAO)	1.38	17 x 11 x 13	45.1	20	15.4
TOTAL	2.19		80.1	83	74.9

3.5.4 Experiment Physical Parameters - The physical parameters of the major components of this experiment are tabulated in Table 3.5-2. The parameters for the star mapper are estimates. The master reference parameters of the star tracker used for Phase II of the experiment are based on the OAO star tracker.

3.5.5 Data Parameters - Table 3.5-3 is a tabulation of the experiment parameters to be telemetered for this experiment. Those signals listed as master reference are not required for the Phase I experiment. The relative need for the parameter is indicated in the desired or essential columns. Additional test points may be added for real-time transmission to aid in trouble and failure analysis.

The information required by Phase I of the experiment is approximate vehicle attitude and the star mapper output signals. Phase II requires a precision measure of the vehicle attitude.

3.5.6 Vehicle Orbit and Attitude Requirements - In order to obtain the maximum usefulness and in order to minimize variables, various orbital parameters of

TABLE 3.5-3
STAR RECOGNITION EXPERIMENT
DATA PARAMETERS

DATA POINT	SIGNAL FORMAT	RANGE OF PARAMETER	SAMPLE RATE (FREQ. RESP.)	ACCURACY	RECORD	ESSENTIAL	DESIRABLE	CONTINUOUS
Star Mapper								
Reticle Position Pulse	Digital			± 1%	X	X		X
Photomultiplier Output	Digital			± 1%	X	X		X
Input Power	Analog			± 2%	X		X	
Temperature	Analog			± 5%	X		X	
Master Reference (Phase II)								
Gimbal No. 1					X	X		
Gimbal No. 2					X	X		
Input Power					X		X	
Temperature					X		X	
Miscellaneous								
Time Reference	Digital	19 Bits		± 1 Bit	X	X		
Vehicle Attitude (3)					X	X		
Vehicle Rates (3)					X	X		
Spacecraft Power Buss					X		X	

the vehicle must be controlled. The most desirable orbital parameters are tabulated in Table 3.5-4. A discussion of the effect of variation of each of the parameters is contained in the following paragraphs:

Altitude - The specific altitude for this experiment is not critical. The indicated 100 to 1000 nautical miles is a desirable range because (a) it is a good range for ground tracking and data recovery, and (b) it provides a test of a star mapper in the altitude range most commonly used.

Eccentricity - Nearly circular orbits are desirable but not essential for this experiment if the vehicle is earth oriented. The greater the ellipticity of the orbit, the higher the rotational rates at apogee and hence the higher the error in performance of the star mapper due to its time constant. If the vehicle is inertially referenced, the ellipticity is not critical.

Inclination - An orbital inclination of 70° or greater is indicated in order that the star mapper performance may be more thoroughly evaluated. Star

TABLE 3.5-4

STAR RECOGNITION EXPERIMENT ORBIT AND ATTITUDE REQUIREMENTS

PARAMETER	NOMINAL VALUE
Altitude	100 to 1000 Nautical Miles
Eccentricity	Less than .05 for an earth oriented vehicle.
Orbit Inclination	Greater than 70°
Vehicle Stabilization	Less than 2° deadband Rates less than .1°/sec. Alignment to orbit plane better than 1° Earth or inertial orientation.

distribution charts indicate maximum density near the galactic equator with minimum density near the galactic poles. It is important that the star mapper be evaluated over the entire range of stellar density distribution.

Stabilization - Small deadbands and low rates are very important for this experiment. Using the time constant of 1 second to measure the star coordinates, an error analysis indicates that a vehicle rate of 0.1°/sec introduces an RMS error of 1.05 arc min in the measurement. Inertial referencing of the vehicle eliminates the vehicle rotation error indicated in Table 3.5-1; however, a compensation can be added in the computer to reduce the error if earth orientation is used.

3.5.7 Equipment Support and Data Handling Requirements - In order to perform this experiment, support equipment on the carrier vehicle is required. The signal conditioning, programming, master reference and environmental control requirements are discussed below. The data handling requirements are general, and Appendix B contains a detailed discussion of Data Handling techniques.

Programmer and Time Reference System - This unit contains the necessary relays, electronics, timers and actuators to provide the following functions:

a. Turn the experiment on and off at predetermined time intervals or upon ground command.

b. Provide a precision time reference required for data reduction.

c. Provide, in Phase II, the necessary search logic and commands for the reference star tracker.

d. Provide the logic necessary to turn the recorder on and off at predetermined times and as a function of whether or not the star tracker is tracking.

Master Attitude Reference - Phase II of the experiment is to evaluate the accuracy of the star mapper. To evaluate the accuracy, a precision reference system will be required. Appendix A contains an analysis of master reference systems with varying degrees of precision. It is suggested that a modified OAO star tracker be utilized for this experiment.

Environmental Control - A thorough analysis of the heat loads and vibration characteristics of the carrier vehicle should be performed prior to finalizing the design of the star mapper. Generally speaking, a design to withstand a temperature range of $0^{o}F$ to $150^{o}F$ and 9 g's RMS vibration is within the state-of-the-art and will not cause undue hardships in the designs of the hardware.

Signal Conditioning - To provide a signal format which may be readily recorded or transmitted, signal conditioning is required. The pulse outputs from the Nortronics star mapper could be processed in a manner similar to the mirror position signal discussed in Paragraph 2.4.7; that is, the pulses could be fed into an automatic reset decimal counter which would give a digital output which is a function of the time interval between pulses. Continued analysis of the signal processing technique should be performed throughout the mapper design phases.

Mounting - It is necessary to mount the star mapper in such a manner that an unobstructed field-of-view is available. The recommended star mapper requires a 45° conical field-of-view with the center line of the cone out of the orbit plane at an angle of 20° with respect to the pitch axis. Other star mappers will have different requirements. In all cases, precision alignment with respect to the vehicle axes and the master reference system will be required.

Electrical Energy Requirements - The total energy required for the proposed 3 days experiment is about 500 watt-hours. The master reference for Phase II will require an additional 500 watt-hours of energy.

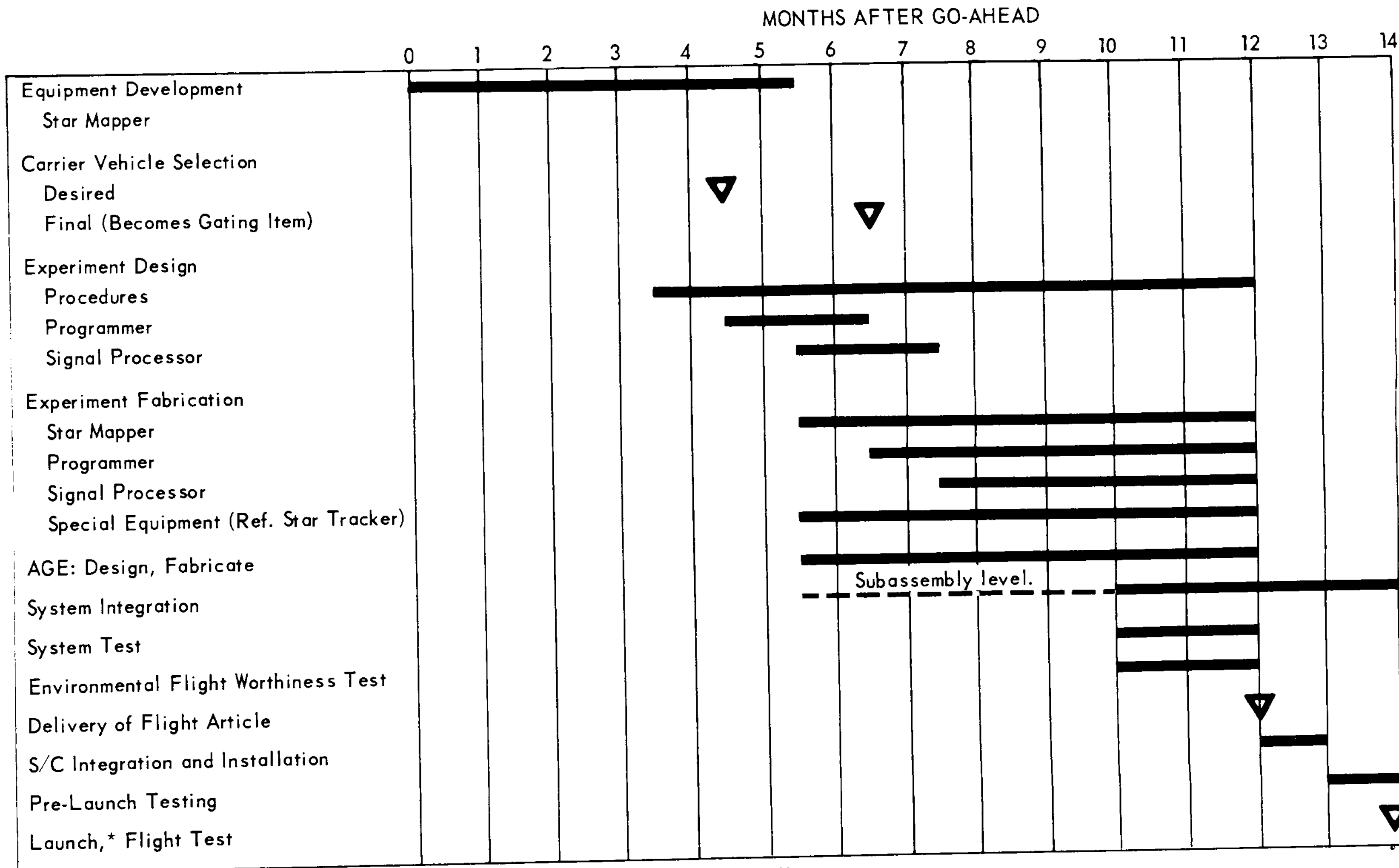
Ground Command Requirements - In order to provide override capability, update or revise the operating sequence, and to update the time reference system, ground commands are required. In addition, ground commands are utilized to turn the experiment on and off and to retrieve the stored data.

The five or six required commands will not require an increase in the ground station equipment but will require that a command receiver be an integral part of the carrier vehicle.

3.5.8 Experiment Flight Test Plan - Figure 3.5-3 is a flow diagram of the experiment development plan. The schedule is an estimate based on the required development, integration testing and flight time and is included for planning purposes. The major gating item in the schedule is the star mapper development time. Failure to select a carrier vehicle early in the program could also cause a schedule delay.

Flight Test Sequence - It is not necessary to operate the experiment continuously while in orbit. In order to minimize the recorder size and yet obtain a large coverage of the stellar field, the experiment is operated in on-off cycles. The suggested sequence would be as follows:

STAR RECOGNITION EXPERIMENT DEVELOPMENT PLAN



*Depends upon carrier vehicle – may vary from 1 month to 8 months after installation.

FIGURE 3.5–3

a. After orbital insertion and vehicle stabilization the experiment is turned on.

b. (Phase I) The star mapper outputs, vehicle attitudes and time reference signals are recorded for 2.5 minutes.

(Phase II) After the master reference star tracker acquires star lock-on the star tracker outputs, the star mapper outputs, time reference and other selected parameters are recorded for 2.5 minutes.

c. The recorder is operated for 2.5 minutes then turned off for 10 minutes.

d. Steps (c) and (d) are repeated for 4 orbits.

e. The experiment is turned off for 4 orbits then steps (b) through (d) are repeated.

f. Steps (d) and (e) are repeated three times.

This cycle or sequence will provide 350 minutes of data in 40 orbits which is a statistically good sample of most of the celestial sphere.

Post Experiment Data Reduction - Data reduction and analysis would proceed as follows:

a. Through use of the star mapper outputs, compute the vehicle attitude. This will require a computer with stored star coordinates.

b. Through use of time correlation of known vehicle orbit coordinates and approximate vehicle attitude, evaluate the data of step (a). Determine the percentage of time that star recognition is accomplished by the mapper-computer combination.

c. For Phase II, compare the star mapper-computer vehicle attitude with the master reference measured attitude.

d. Perform a statistical analysis of the performance characteristics.

REFERENCES

1. Canopus Tracker Built by Santa Barbara Research for Surveyor - See "Canopus Star Sensor will Provide Method to Correct Surveyor Course" Missiles and Rockets, 6 July 1964, Pages 36, 37.

2. D. Egger and P. Billington, "A Star Identification System", Nortronics Publication NORT 62-57, February 1962.

3. J. H. Flink, "Star Identification by Optical Radiation Analysis", IEEE Transactions Volume ANE-10, September 1963, No. 3.

4. John E. Abate, "Star Tracking and Scanning Systems, Their Performance and Parametric Design" IEEE Transactions Volume ANE-10, September 1963, No. 3.

5. Floyd V. McConless, "A Systems Approach to Star Trackers" IEEE Transactions Volume ANE-10, September 1963, No. 3.

6. S. Moskowitz, "Instrumentation for Space Navigation", IEEE Transactions Volume ANE-10, September 1963, No. 3.

7. Nortronics Proposal NORT 63-328B, "An Unsolicited Proposal for a Star Field Scanner for Space Vehicle Attitude Sensing", Submitted to the Office of Research Grants and Contracts, NASA, November 1963 (Revised January 1964).

3.6 Small Impulse Devices

3.6.1 Objectives - The objective of this experiment is to determine ignition characteristics, average pulse size, and effective I_{sp} of a small impulse reaction device, and expose any problems caused by the orbital environment.

3.6.2 Background - Reaction jets are used in a majority of space vehicle attitude control systems, either as the primary torque device or as a momentum dumping device. They operate in conjunction with switching logic (supplying fixed thrust when errors exceed a fixed minimum) rather than an analog mode (supplying thrust proportional to errors). With such a device, it is not possible to compensate exactly for the small torques introduced by the environment and reduce the vehicle rates to zero. Instead, a limit cycle is established in which the vehicle motion remains within fixed position and rate errors. One of the major considerations in reaction control design is to optimize the limit cycle, i.e. produce the smallest possible rate and attitude deadbands with the least fuel expenditure.

The rate deadband can never be less than one-half the rate change produced by the minimum impulse. This emphasizes the need for smaller impulse reaction devices. One approach to their design is scaled-down versions of existing systems, e.g., cold gas, monopropellant hot gas, and hypergolic systems. Another approach is to develop new types of jets, six of which are discussed below and described in Table 3.6-1. These jets are being designed for minimum impulses of less than 10 millipound-seconds.

Resistance Jets - The resistance jet is an outgrowth of the cold gas jet. Of the basic gas jets, the cold gas system can be designed to produce the smallest impulse. Storage of the gas rapidly becomes prohibitive from a size and weight standpoint as the total impulse increases. The resistance jet is a cold gas system with a heated nozzle which increases the specific impulse by a factor of 8 to 10

TABLE 3.6-1

CANDIDATE REACTION JETS

ITEM	WEIGHT (LB.)	VOLUME (FT.^{3})	MINIMUM PULSE WIDTH (SEC.)	MINIMUM THRUST LEVEL (LB.)	MAXIMUM THRUST LEVEL (LB.)
Resistance Jet				10^{-2}	1
Thrust Chamber & Valves	1.3	.05			
Propellant Supply	1.5	.05			
Sublimation Jet	10	.1		10^{-4}	10^{-1}
Ion Jet				10^{-6}	10^{-4}
Plasma Jet			.002	10^{-6}	10^{-4}
Thrust Chamber	6.5				
Propellant Supply	0.6				
Detonation Jets			.0004	1	10
Thrust Chamber & Valves					
Propellant Supply					
Solid Propellant Jets			.02	1	10
Stepping Mechanism	0.5				
Propellant Supply	0.002/shot				

(Reference 1). Thus, the resistance jet has all of the good characteristics of the cold gas system without the costly storage problem.

Sublimation Jets - Sublimation jets operate on the principle that, under proper conditions of temperature and pressure, certain materials sublime directly from the solid to the gaseous state. The gas is then used (as in cold gas systems) to provide thrust. A system of this type can be controlled by either pressure or temperature. Pressure control is accomplished by placing the solid in a chamber whose only exit is a valve which will vent the gas to space. The operating cycle consists of two steps: beginning with an evacuated chamber, sublimation commences and continues until sufficient gas pressure exists to halt the process; opening the valve evacuates all, or part, of the gas and sublimation recommences. Temperature control requires a material which needs both a vacuum and heat to sublime. The material is stored in a tank which has one opening in the form of a nozzle. A filament is mounted within the tank. When the filament is turned on, sublimation commences and continues at a steady rate until the filament is turned off. The sublimed gas continually evacuates to space through the nozzle. No moving parts are involved.

Ion Jets - Ion jets function by generating ions, accelerating them through an electric field and expelling them. This device has been considered primarily as a propulsion system for long term space missions because of its high specific impulse. This inherent advantage makes it attractive as a control device. However, power requirements for ion jets are high and these units are not readily adaptable to multiple starts. In addition, ion jet development has encountered a problem known as "beam neutralization". If the ions alone are expelled, the vehicle becomes negatively charged, which is undesirable. Electrons must be placed in or near the ion beam to neutralize it. Several methods of doing this are under investigation.

Plasma Jets - Plasma jets are similar in some ways to ion jets. Parallel plates are capacitively charged in a vacuum. A gas is introduced between them, which quickly breaks down as the capacitor discharges through the plasma which is formed. A magnetic field is set up by the current flow. The interaction between the electric and magnetic fields forces the plasma out of the vehicle. The advantages of plasma jets over ion jets are higher efficiency, faster response time, and a neutral plasma. The disadvantage is in the requirement for a gas supply system.

Detonation Hypergolic Jets - Detonation hypergolic jets are bi-propellant jets which operate on a pulse basis. Minute quantities of the propellants are metered into separate storage chambers, then simultaneously injected into the thrust chamber. The sudden hypergolic reaction resembles an explosion of very small magnitude. Major development problems requiring solution are the propellant measuring system and the injection system. Both of these have a direct effect on the impulse size.

Solid Propellant Reaction Jets - This device is best described by comparing it to a toy cap pistol. Solid propellant charges, each with its individual nozzle, are encapsulated in a strip of material. Upon command, a charge is stepped into firing position and combustion occurs. Although the charge feed mechanism is more complex than other systems, this technique will provide high consistency in pulse size. The individual nozzles remove the problem of nozzle deterioration.

No orbital flight experience is available on the performance of the low level reaction jets with the exception of ion engines. Several flights were made on different types of ion engines to measure their performance as a propulsion unit and to determine the scope of the beam neutralization problem. The effects on beam neutralization of a space chamber acting as a getter was unknown. The flight test results indicated that these effects were not great. Resistance jet efficiency is dependent on thermal isolation of the heater and operate most efficiently at pressures less than 10^{-9} torr. Because of the minute force levels of the reaction jets being masked by earth produced torques and forces, chamber pressure is used during ground test to measure performance. Nozzle design and gas flow characteristics, in addition to chamber pressure, determine the actual thrust level.

Orbital testing of small impulse reaction jets is desirable because knowledge of their flight performance is lacking and earth produced forces or torques mask the results of ground tests performed on devices with less than 10^{-4} lbs. of thrust.

3.6.3 Functional Description - This experiment is designed to accommodate four of the six jet designs discussed in paragraph 3.6.2. The low thrust levels of ion and plasma jets will require special design consideration. A typical experimental system will consist of the components shown in the Functional Block Diagram (Figure 3.6-1). A typical Thruster and Gimbal Assembly, containing the thrust chamber and propellant supply, is shown in Figure 3.6-2. If it proves desirable to test more than one of the candidate thrusters, multiple thruster and gimbal packages or mounting multiple thruster assemblies on one gimbal could be employed.

SMALL IMPULSE DEVICES EXPERIMENT FUNCTIONAL BLOCK DIAGRAM

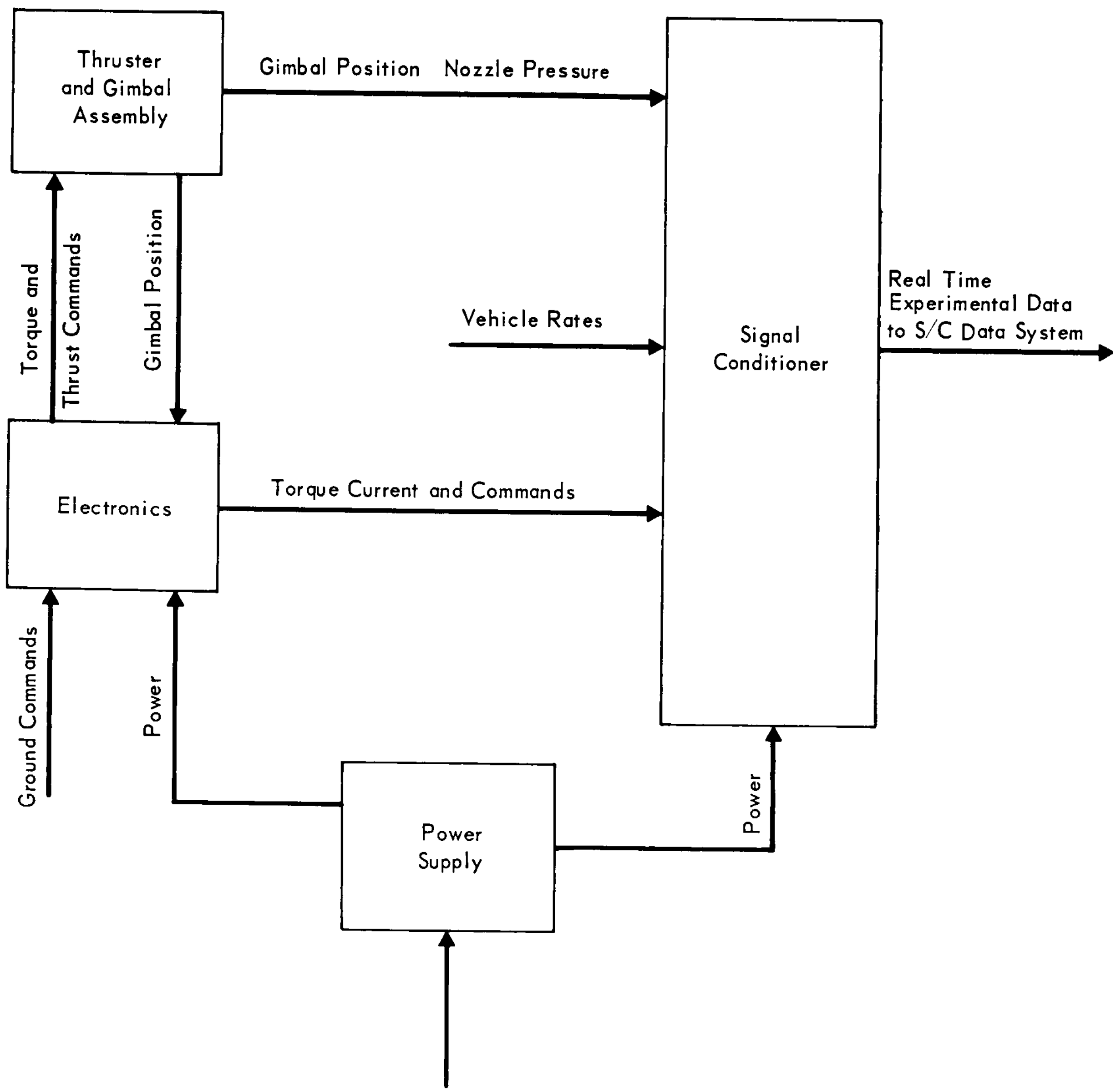


FIGURE 3.6-1

TYPICAL THRUSTER AND GIMBAL ASSEMBLY

SMALL IMPULSE DEVICE

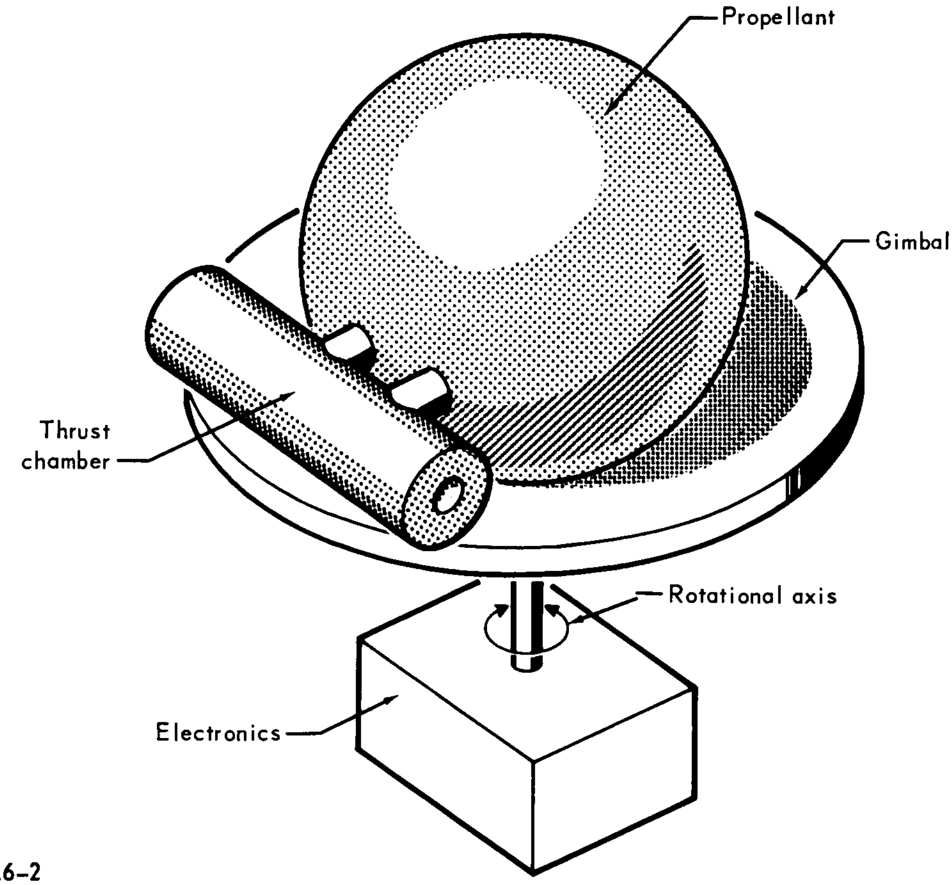


FIGURE 3.6-2

Thrust determination is primarily accomplished by measuring the current required by a torquer to hold the gimbal immobile against the thrust of the reaction device. This should provide an accuracy within five percent. Secondary measurements will be made of nozzle pressures.

Experiment Equipment - Experiment equipment will consist of the gimbal, the thrust chamber and its propellant, and the electronics. The gimbal is a rotating table on which the thrust chamber and its fuel supply are mounted. The thrust chamber is mounted so that its thrust vector is perpendicular to but offset from the gimbal axis. The gimbal and its equipment are mass balanced about the rotational axis.

The gimbal control is a closed loop servo consisting of a position pickoff, an amplifier and a torquer. Its function is to hold the gimbal in a fixed position with respect to the vehicle. The gimbal position pickoff provides both a feedback signal to the control electronics and a monitor signal to the data handling system. The position signal is amplified and applied to the torquer, which provides the torque to balance the jet thrust torque. This approach will be capable of measuring 10^{-4} pounds of thrust at a three-inch moment arm. Gimbal freedom is limited by stops to a small angular travel which will permit the use of flexible electrical leads instead of slip rings to provide electrical contact between the electronics and gimbal mounted equipment.

Test Method - The short duration of a test cycle enables this experiment to be conducted while the carrier vehicle is in contact with a ground station. Upon ground command, electrical power will be applied to the experimental equipment. After a short warm-up time, another ground command will initiate the test. The reaction jet will commence a series of impulses which will be terminated by ground command. Power will then be removed from the experimental equipment. This test cycle will then be repeated until the propellant supply is exhausted.

3.6.4 Experiment Physical Parameters - Table 3.6-2 defines the size, weight and power requirements of the experimental components. Sufficient allowances have been made for the thruster assembly to permit the choice of any of the candidate systems described in Table 3.6-1.

3.6.5 Data Parameters - Table 3.6-3 contains a list of all experimental parameters which shall be monitored. The nature of the thrust chamber parameters will be determined by the type of thruster being tested.

TABLE 3.6-2

SMALL IMPULSE DEVICES EXPERIMENT PHYSICAL PARAMETERS

EQUIPMENT	SIZE		WEIGHT (LB.)	POWER	
	VOLUME ($FT.^3$)	L-W-H (IN.)		PEAK (WATTS)	NOMINAL (WATTS)
Experimental					
Thruster Assembly and Propellant	0.296	8 x 8 x 8	10	5*	1*
Gimbal	0.062	6D x 1	5		
Electronics	0.010	3 x 3 x 2	0.5	10*	5*
Support					
Signal Conditioner	0.004	3 x 2 x 1	0.1	5*	1*
Power Supply	0.010	3 x 3 x 2	0.9	28	11
Total	0.382		16.5	28	10

*Supplied by Power Supply

3.6.6 Vehicle Orbit and Attitude Requirements - The only constraint made by this experiment is on the vehicle angular acceleration. This must be kept below 0.1 degree per second per second. The most simple solution to this requirement will be to disable the vehicle attitude control system during the test cycle.

3.6.7 Experiment Support and Data Handling Requirements - The data handling requirements discussed in Appendix B are applicable. The transponder and the recorder are not required since vehicle position is not necessary and all data will be monitored in real-time. A master reference is not required and no unique problems are expected in the areas of signal conditioning and environmental control. The total energy, mounting and command requirements are described below.

Mounting - Care must be exercised in locating the thruster and gimbal assembly so that the jet exhaust does not impinge on the vehicle surface. In addition, alignment of the thrust vector through the vehicle center of mass will limit the

TABLE 3.6-3

SMALL IMPULSE DEVICES EXPERIMENT DATA PARAMETERS

DATA POINT	SIGNAL FORMAT AND FREQUENCY	PARAMETER RANGE	SAMPLE RATE (/sec.)	ACCURACY	ESSEN-TIAL	DESIR-ABLE	REAL TIME
Gimbal Position		±5 deg.			X		X
Torquer Current					X		X
Thrust Chamber	Analog, DC			1% F.S.	X		X
Thrust Chamber					X		X
Thrust Chamber			50		X		X
Vehicle Roll Rate						X	X
Vehicle Pitch Rate	Analog, 400 cps	± 1.0°/sec.		±0.01°/sec.		X	X
Vehicle Yaw Rate						X	X
Power Command	Analog, DC	on/off	0.1			X	X
Thruster Command						X	X

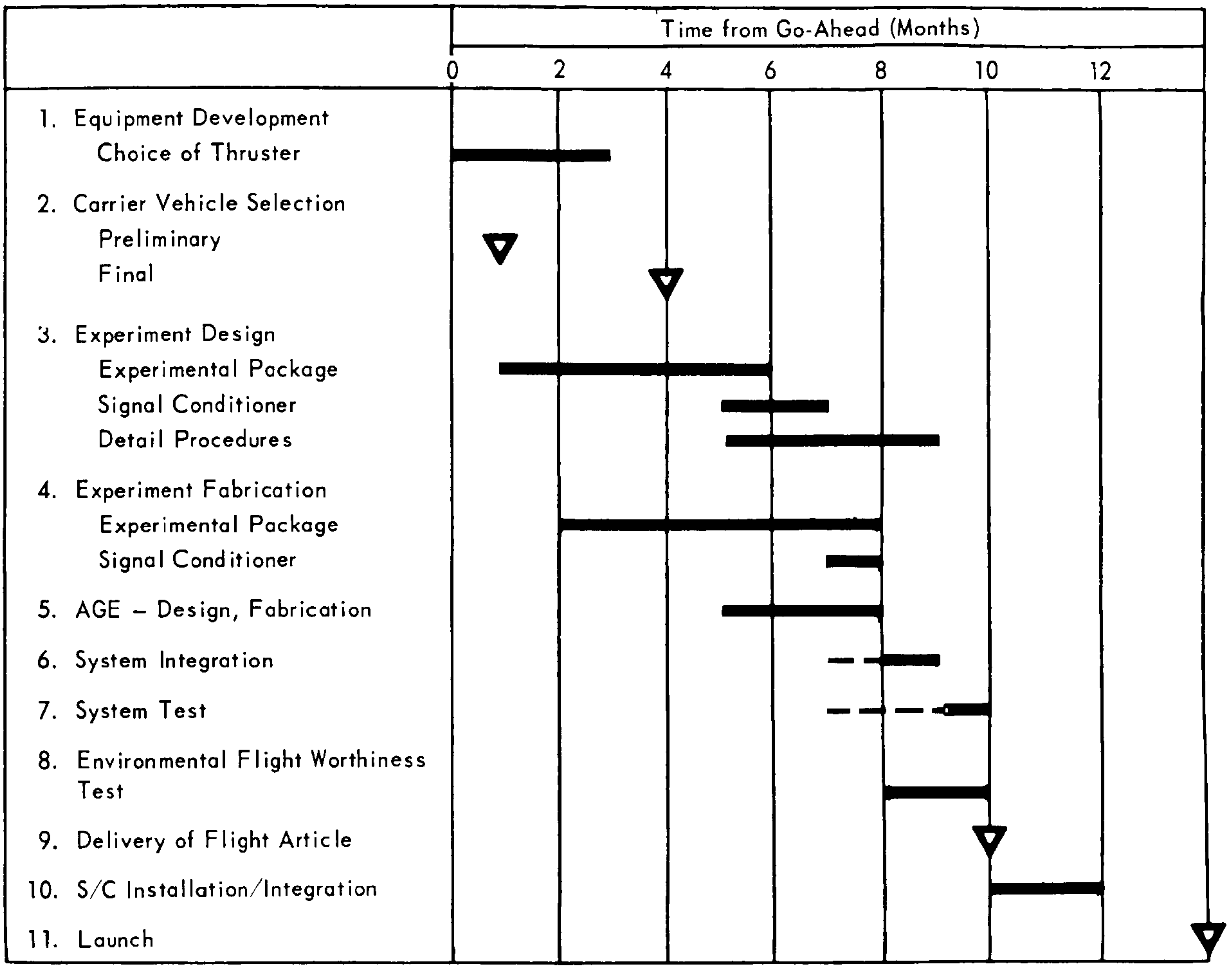
effect on vehicle motion. The angular acceleration induced will be equal to the thrust times the moment arm divided by the vehicle inertia. If this acceleration is excessive, an alternate test design would use two opposed thrust chambers which would pulse in turn.

Commands per Experiment Sequence - Two ground commands are required to perform the experiment. The "Power" command applies and removes power from all experimental packages. The "Thrust" command initiates and terminates the jet thrust.

Electrical Energy Requirements - The size and weight allowance for propellant will supply fuel for a minimum of ten test cycles. The total energy required for the ten cycles will be approximately 14 watt-hours.

3.6.8 Experiment Flight Test Plan - The development plan for the program is detailed in Figure 3.6-3. Gating factors are the choice of thrusters, carrier

SMALL IMPULSE DEVICES EXPERIMENT DEVELOPMENT PLAN



*Depends on Carrier Vehicle. May vary from 1 to 8 months after installation.

FIGURE 3.6-3

vehicle final selection and experimental equipment design and fabrication. The flight test sequence (Figure 3.6-4) utilizes the following steps all of which occur during one period of contact with a chosen ground station.

a. Power is applied to all experimental equipment. Two minutes warm-up time is allowed.

b. Thrust impulses are initiated and continue for two minutes.

SMALL IMPULSE DEVICES EXPERIMENT
FLIGHT TEST SEQUENCE

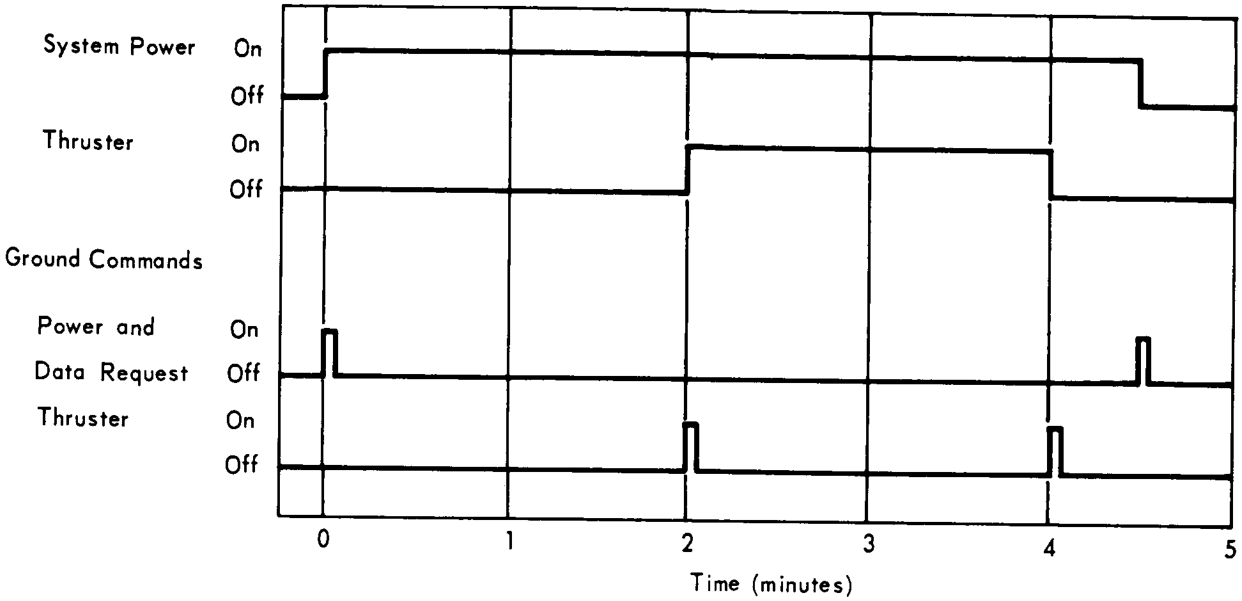


FIGURE 3.6-4

c. Thrust impulses are terminated.

d. After thirty seconds, power is removed from the experimental equipment.

REFERENCES

1. Resistance Jet Attitude Control System, General Electric Co., LMED, Report LMEJ 7154-2.

3.7 Optical Windows and Mirrors

3.7.1 Objectives - The objective of this experiment is to quantitatively measure the optical degradation which occurs in windows and first surface mirrors when they are exposed to the space environment.

3.7.2 Background - A majority of the optical tracking systems in use or under design are sealed units requiring optical windows. Most of the remainder employ first surface mirrors to reflect the light signals to the sensor.

Windows must be designed with several important qualities which affect the value of the light signal presented to the sensors. Surface flatness and a parallel relationship between the front and back surfaces affect the distortion of the image. The transmissibility of the window in the desired frequency spectrum affects the signal level received by the sensor. Existing manufacturing techniques are quite capable of producing windows which exhibit high quality in these areas in the ambient ground environment. In space the windows are subjected to long time exposure to such environmental factors as radiation, temperature shock, meteorites, subliming and outgassing materials, and jet exhausts.

Ground testing of optical devices has limited usefulness. Radiation testing is possible with good quantitative results. Inside a vacuum chamber the lens is exposed to a radiation source. The material damage can be closely correlated to type of radiation and length of exposure. To some extent, sublimation coating tests can be performed on the ground. Realistic tests of coating from outgassing and jet exhausts, and of meteorite impingement are not presently feasible primarily because insufficient quantitative data is available for test definition. The degradation effects of the orbital environment are shown in Table 3.7-1. The necessity of tests under a realistic combination of all these environments and the difficulty of simulation make an orbital test advantageous.

TABLE 3.7-1

SPACE ENVIRONMENT DEGRADATION

ENVIRONMENT	DEGRADATION
Radiation	Molecular changes from radiation bombardment.
Temperature Shock	Warpage
Meteorites	Scratches and pits in the surface.
Outgassing and Subliming Materials	Coating which impedes the passage of light.
Jet Exhausts	same as previous

The choice of window materials is dependent upon the light spectrum to be transmitted. Normal optical quality glass will transmit a broad spectrum running from the near ultraviolet to the near infrared (approximately 0.32 micron to 2.2 micron). Germanium or special composition windows are used for the transmission of the infrared spectrum.

A typical first surface mirror consists of a copper or beryllium plate with a deposited aluminum surface which is then coated with silicon monoxide to prevent oxidation of the aluminum. Such a mirror can be used to give high reflectance over a spectrum running from the infrared through the ultraviolet regions. In the space environment mirrors are subject to the same degradation effects as windows.

Factors which must be considered before an experiment configuration can be defined are the reference light source, carrier vehicle orientation, and multiplicity of experimental packages. Logical light sources to be used as a reference are the sun and artificial sources included in the experimental package. The major advantages of using the sun are simplicity and reliability of the experimental package. However, the reliability of the experiment package may be offset by the complexities of programming the vehicle solar orientation at required times. The major disadvantages of using the sun as a reference source are the vehicle pointing accuracy and stabilization requirements. Pointing accuracy would be critical during the periods when data is being recorded (the light path length through the window will directly affect the signal level). Use of an artificial light source offers independence of the experiment from vehicle orientation, thus the test becomes adaptable to many different vehicles.

Mounting one experimental package on the sun side and another on the dark side of a solar oriented vehicle or a solar paddle provides a very diagnostic test. Radiation damage and temperature shock are more severe on the sun side experiment while the dark side experiment will be more susceptible to fogging or coating. Although it will be impossible to achieve an exact measure of the degradation caused by any one environment, some measure of comparative effects may be possible. Mounting one experiment on the sun side and one on the dark side tends to rule out the use of the sun as the reference light source.

Based on these factors, a window experiment package and a mirror experiment package will be designed, each of which will provide its own artificial radiation sources. If the carrier vehicle is scheduled to be solar oriented, such as OSO, two of each will be carried and mounted as described above. With any other orientation, one package of each type will be tested. In either case, one set of supporting equipment will be employed.

The nature of the test is such that consideration should be given to the possibility of recovering the mirror and window packages intact. This would allow a much more thorough determination of the amount and type of degradation which has occurred.

OPTICAL EXPERIMENT FUNCTIONAL BLOCK DIAGRAM

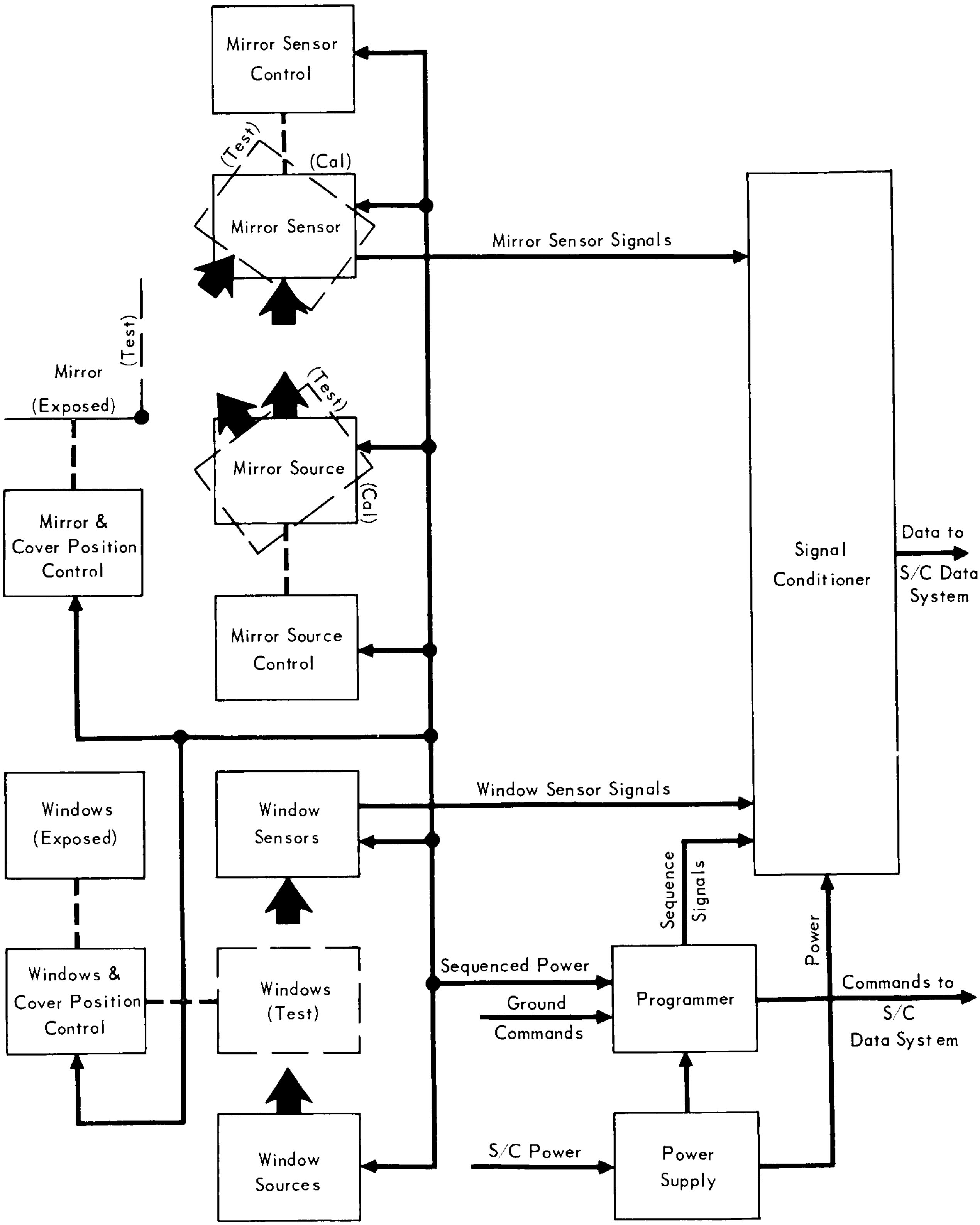


FIGURE 3.7-1

3.7.3 Functional Description - A functional block diagram of the experiment is shown in Figure 3.7-1. Experiment equipment consists of the windows and mirrors and their associated sources and sensors. The programmer, signal conditioner, and power supply are experimental support equipment. The experimental packages are designed in accordance with the following ground rules.

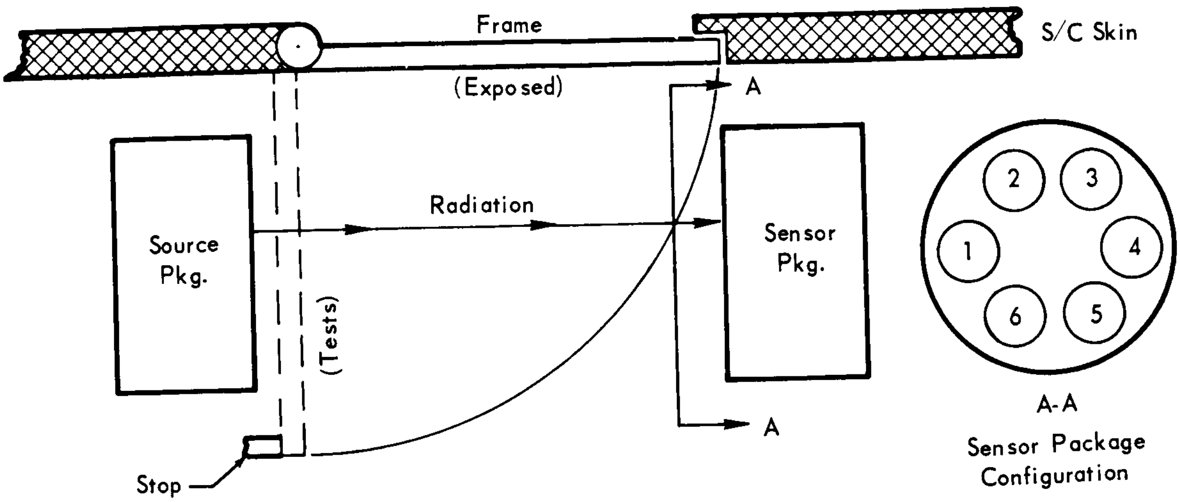
a. Each window and mirror package is provided with covers to protect the sources and sensors from radiation and meteorite damage.

b. During exposure periods, windows, mirrors and covers are mechanically locked in position and all power removed from the experimental system.

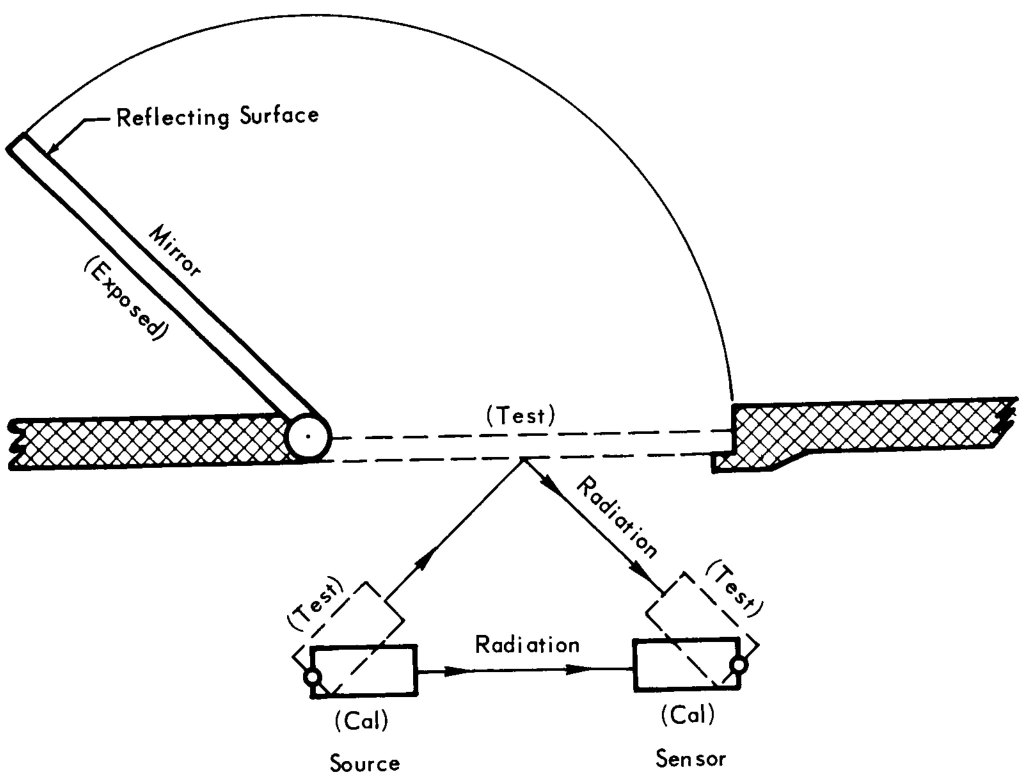
c. Basic design of the equipment will be made independent of the possible sources, sensors, windows and mirrors.

Windows and Covers - Six circular windows are mounted in a movable frame which exposes the windows to the space environment or places them inside the vehicle for exposure to the test source. The choice of six windows is arbitrarily based on achieving a compact mechanical design. When exposed to the space environment, the window frame is flush with the vehicle skin as shown in Figure 3.7-2a. For test purposes, the frame will rotate inward placing the windows between the light sources and sensors. During the test, the frame is held tightly against a stop which is adjusted on the ground. The frame is adjustable in a manner which places each window successively between each source and sensor combination. The characteristics of a group of candidate window materials are shown in Figure 3.7.3. As an alternate design, one point source, such as a special tungsten source or a mercury or zirconium arc lamp, is operated in conjunction with a series of filters to provide the desired spectral characteristics. This design will require less power but require a longer test cycle which will be more complex to program.

OPTICAL EXPERIMENT MECHANIZATION



(a)



(b)

FIGURE 3.7-2

CANDIDATE WINDOW MATERIALS (Reference 1)

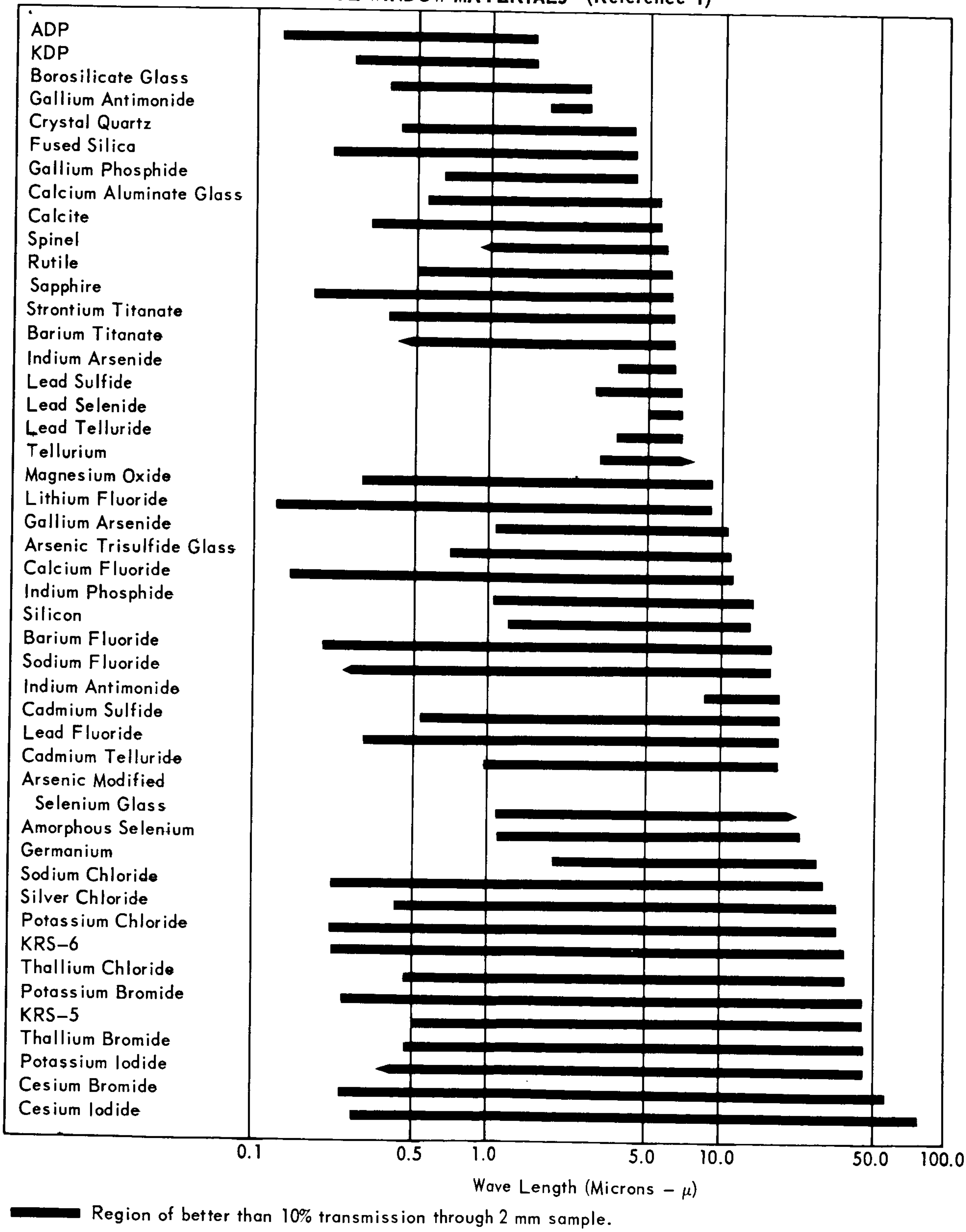


FIGURE 3.7-3

Window Source and Sensor Package - This package contains the source and sensor combinations which provide radiation of the chosen frequencies. The sources are filament bulbs, Osram (spectral) lamps and/or black body sources which are filtered to provide radiation in the six desired bandwidths to be chosen at a later time. The sensors are thermistor bolometers. The sources are collimated to provide maximum radiation for the available power. The amount of collimation depends upon the image size and/or total light required at the sensor. The optical center lines of the source and the sensor will be colinear. The outputs from the package are signal levels from the sensors which will be a measure of the quantity of light reaching them.

Mirror and Cover - The mirror is first surface mirror whose composition will be chosen from the list of candidates in Table 3.7-2. The mirror frame has two positions, exposed and test. In the test position, the mirror is flush with the vehicle surface and held tightly against the stop. Stop adjustment is accomplished during ground testing. Mirror position for both the exposed and test conditions is illustrated in Figure 3.7-2b.

TABLE 3.7-2
CANDIDATE MIRRORS

BASE	REFLECTIVE COATING	PROTECTIVE COATING	BANDWIDTH
1. Beryllium	Aluminum (Al)*	Silicon Monoxide (SiO)	Broad
2. Beryllium	Aluminum (Al)*	Magnesium Fluoride (M_gF_2)	Broad
3. Glass	Aluminum (Al)*	Silicon Monoxide (SiO)	Broad
4. Glass	Aluminum (Al)*	Magnesium Fluoride (M_gF_2)	Broad

*Copper (Co) or Gold (Au) for special purposes

Mirror Source and Sensor - The mirror source and sensor are designed in a manner similar to that of the window source and sensor package. Each unit has two operating positions, calibration and test. The optical centerline of the two units are colinear during calibration. During test, the units are aligned so that their optical centerlines meet at the mirror surface. Both of these positions are achieved by torquing the units against stops which are adjusted on the ground. The source is collimated to the requirements described above. The sensor emits signal levels corresponding to the radiation received.

Position Controls - The position controls are integral with the other packages. These include the solenoids and motors required to position all moving assemblies.

Test Method - Once each day, a complete test cycle is performed. This consists of calibrating each sensor by measuring the amount of radiation being produced by its source. Then the windows and mirrors are placed in the light path and the sensors are again monitored. Each window is tested sequentially with each source and sensor. The ratio of the test and calibration signals is a measure of window transmissibility or mirror reflectivity. Comparison of this ratio to that of previous tests will define the total degradation which has taken place. The base ratio will be determined prior to the launch of the carrier vehicle.

3.7.4 Experiment Physical Parameters - Table 3.7-3 describes the size, weight, and power requirements for the experimental components. The size and weight figures for the experimental packages include two units of each. If only one unit of each type is used, these are halved, although the power requirement and the estimates for the supporting equipment remain unchanged.

3.7.5 Experiment Instrumentation - Table 3.7-4 contains a list of all parameters which are monitored for this experiment. Recovery of the experimental

TABLE 3.7-3
OPTICAL EXPERIMENT PHYSICAL PARAMETERS

ITEM	VOLUME		WEIGHT	AVERAGE POWER (WATTS)	
	CU.FT.	L - W - H	(LB.)	PEAK	NOMINAL
Experimental					
Window Experimental Package (2)	.063	@ 6 x 3 x 3	5.0	15*	6*
Mirror Experimental Package (2)	.019	@4 x 2 x 2	3.0	12*	1*
Support					
Programmer	.019	4 x 4 x 2	1.0	3*	1*
Signal Conditioner	.004	6 x 1 x 1	.2	4*	2*
Power Supply	.052	6 x 5 x 3	3.9	40	14
TOTAL	.157		13.1	40	14

*Supplied by Power Supply

units from orbit would be desirable to assist in the final analysis of exposure degradation.

3.7.6 Spacecraft Control and Orbit Requirements - The experiment has no firm requirements for either vehicle control or orbit parameters. If a sun oriented vehicle is used, the attitude should be maintained within ± 30 degrees. With any other orientation, there are no requirements. A vehicle with an orbit passing through the Van Allen belt will provide a more severe radiation test, but this fact is secondary compared to the desirability of performing a test between two identical units on the sun-oriented vehicle.

3.7.7 Experiment Support and Data Handling Requirements - The data handling requirements discussed in Appendix B are applicable. Ground tracking aids are not required for this experiment since vehicle orbital position is not important. The design of the experiment support equipment does not appear to offer any problems.

TABLE 3.7-4
OPTICAL EXPERIMENT DATA PARAMETERS

DATA POINT	SIGNAL FORMAT AND FREQUENCY	PARAMETER RANGE	SAMPLE RATE (/SEC.)	ACCURACY	RECORD	ESSEN-TIAL	DESIR-ABLE
Pkg. 1 – Window Sensor No. 1	Analog, DC	0–5 volts	1	±0.05 volts	X	X	
No. 2					X	X	
No. 3					X	X	
No. 4					X	X	
No. 5					X	X	
No. 6					X	X	
– Mirror Sensor					X	X	
Pkg. 2 – Window Sensor No. 1					X	X	
No. 2					X	X	
No. 3					X	X	
No. 4					X	X	
No. 5					X	X	
No. 6					X	X	
– Mirror Sensor					X	X	
Command – Test Pkg. No. 1		Off/On			X		X
No. 2					X		X
– Source Power On					X		X
– Unlock Power On					X		X
– Step Windows					X	X	
Pkg. 1 – Window Position		Limit 1/ Transition/ Limit 2			X		X
– Mirror Position					X		X
– Window Cover Position					X		X
– Mirror Cover Position					X		X
Pkg. 2 – Window Position					X		X
– Mirror Position					X		X
– Window Cover Position					X		X
– Mirror Cover Position					X		X

NOTE: All data points monitored during test periods and on ground command

There is no requirement for a master attitude reference, nor is there an environmental requirement. Each daily test cycle described in Figure 3.7-5 requires 7.5 watt-hours of energy for a monthly total of 225 watt-hours. Mounting and ground commands are discussed below.

Mounting - Mounting of the experimental units must be given special consideration. The units must be mounted so that the windows and mirrors are subject to the space environment in the exposed position. In the case of the solar-oriented vehicle, the two sets of packages should be mounted as nearly as possible on opposite sides of the vehicle. The support equipment may be placed in any non-interfering position within the vehicle.

Ground Commands - The experiment is performed in conjunction with three specific ground commands. "Initiate Test" will result in the experiment being conducted through one complete test cycle. "Dump Data" will result in the transmission of all data stored on-board the vehicle. The "Monitor Data" command will result in the transmission of real-time data. This command serves to turn the data system both on and off. Step-by-step performance of the experiment via ground command is not considered feasible. The added reliability which would come from having a backup capability would be offset by the added complexities of the required ground command receiving equipment.

3.7.8 Experiment Flight Test Plan - The development plan for the program is detailed in Figure 3.7-4. The gating factor in the estimated ten month time base is the six month allowance for design and fabrication of the experimental packages. The flight test sequence is defined in Figure 3.7-5. It involves utilization of the following steps. Each experimental package consists of one window package and one mirror package.

a. Energize Package 1 sources and allow sufficient warmup time.

b. Calibrate Package 1 sensors by monitoring their output while the windows and mirror remain in the exposed condition. The mirror source and sensor will be in the calibrate position.

c. Cycle the Package 1 windows and mirror and the associated covers from the exposed to the test positions. Cycle the mirror source and sensor to the test position.

d. Conduct a test cycle, interposing each window source and sensor combination by each window successively.

OPTICAL EXPERIMENT DEVELOPMENT PLAN

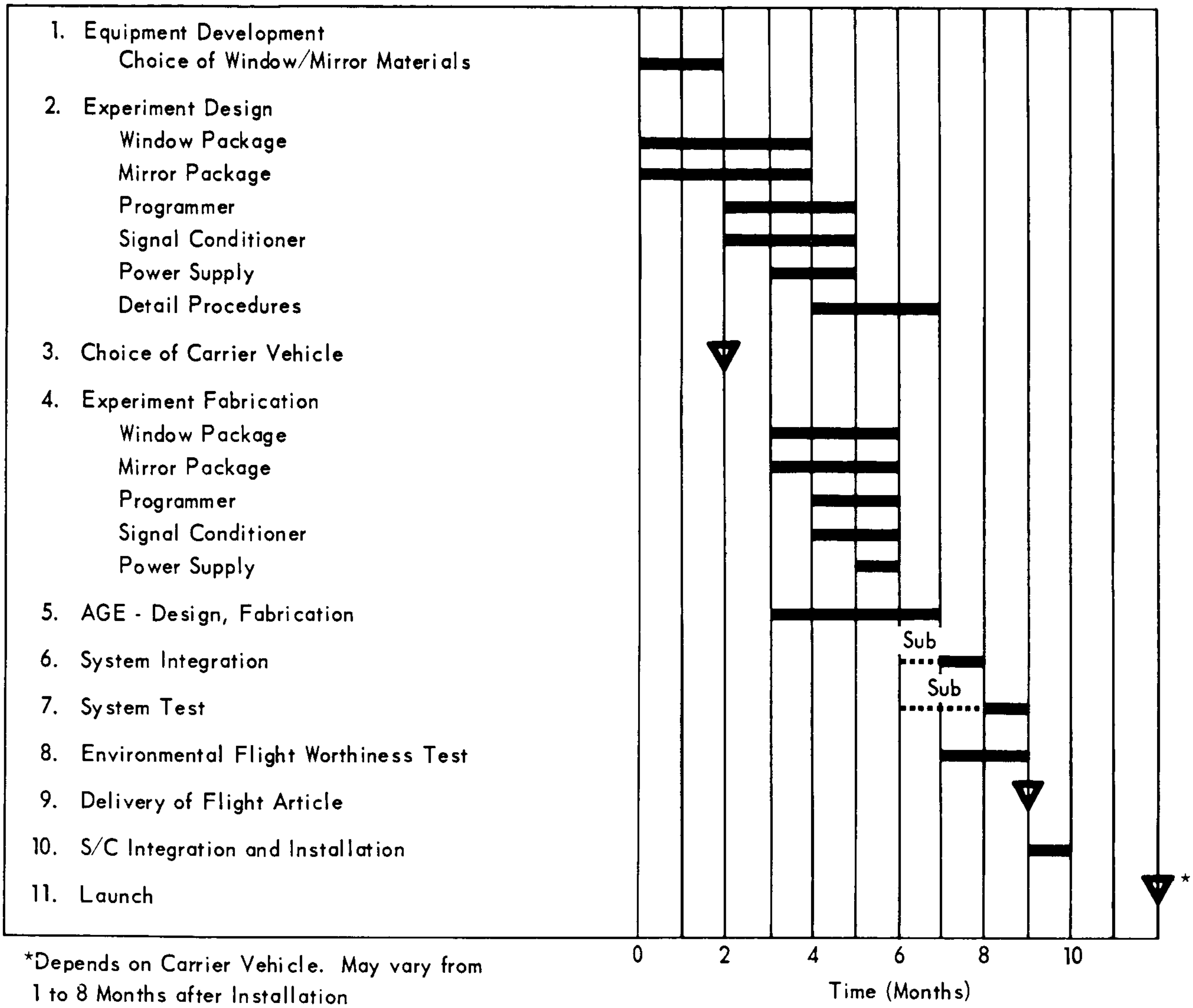


FIGURE 3.7-4

e. Recycle all equipment to the exposed or calibrate position.

f. Repeat steps (a) through (e) using Package 2.

g. Turn off power.

h. Repeat this cycle once per 24 hour period for the duration of the test.

Data obtained from this experiment should be processed into a series of charts. Each chart will detail the performance of one window in conjunction

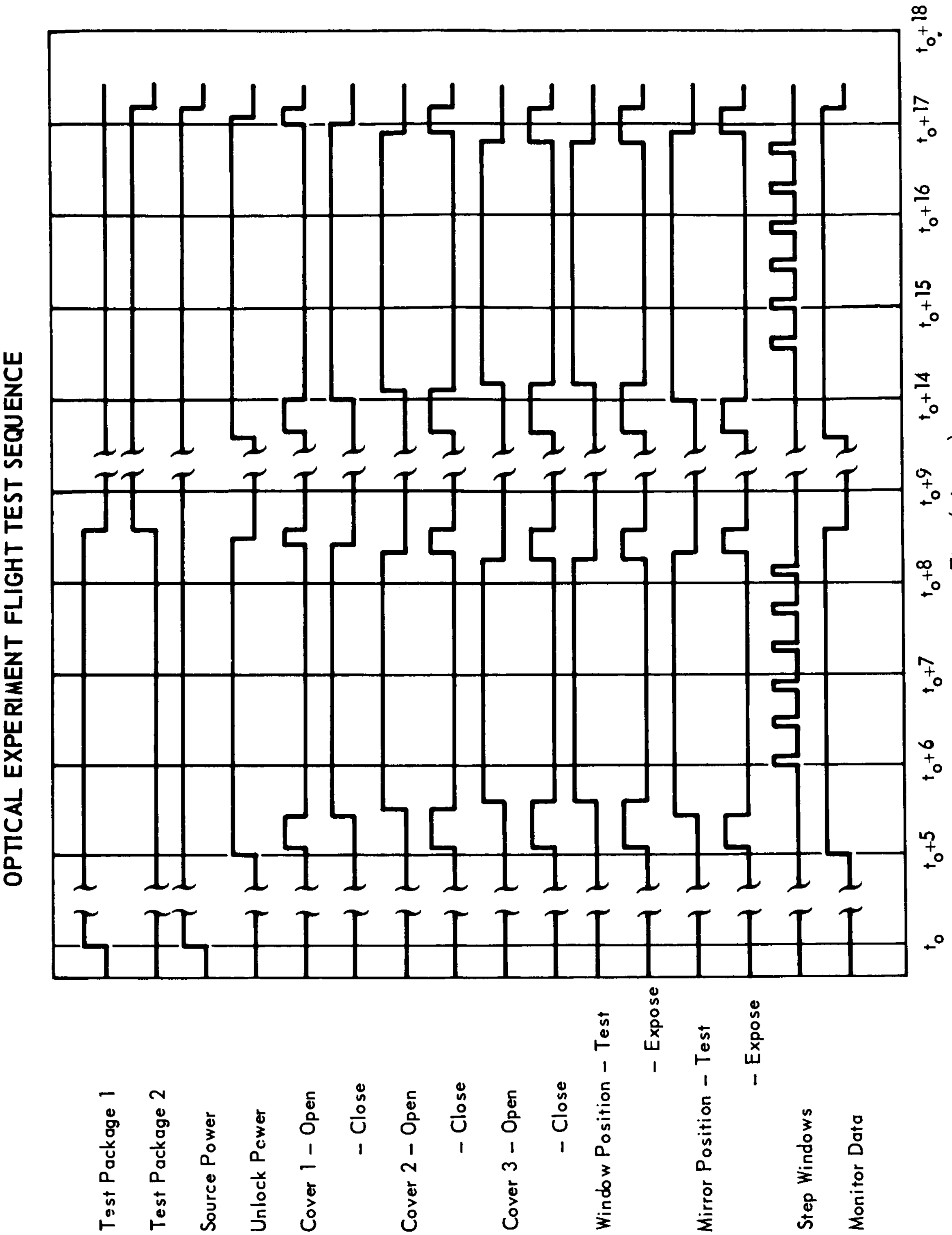

OPTICAL EXPERIMENT FLIGHT TEST SEQUENCE
Test Package 1
Test Package 2
Source Power
Unlock Power
Cover 1 – Open
– Close
Cover 2 – Open
– Close
Cover 3 – Open
– Close
Window Position – Test
– Expose
Mirror Position – Test
– Expose
Step Windows
Monitor Data
t_o
t_o+5
t_o+6
t_o+7
t_o+8
t_o+9
t_o+14
t_o+15
t_o+16
t_o+17
t_o+18
Time (minutes)

FIGURE 3.7–5

with one source and sensor combination (or the mirror with its source and sensor). The absissa of the chart is time from the beginning of the test. The ordinate is the normalized performance, which is defined by the equation:

$$P_N = \frac{A_N/B_N}{A_O/B_O}$$

where: P_N = Performance on the Nth test;

A_N = Sensor signal with window in test position for Nth test;

B_N = Sensor signal during calibration of Nth test;

A_O = Sensor signal with window in test position before initial exposure;

B_O = Sensor signal during calibration before initial exposure.

Treatment of the data in this manner will eliminate errors due to changes in light intensity from the source. There will be six charts for each window, one for each mirror.

It would appear that this experiment would only be the first in a series of tests to find satisfactory window and mirror materials. Subsequent tests could utilize a new set of candidate windows or run a more thorough test on a particularly promising material. In either case, the design of the basic experimental hardware and the support equipment would be complete.

REFERENCES

Infrared Systems Handbook, McDonnell Aircraft Corporation, Technical Memo No. 412-AE-31220, dated 27 December 1963.

3.8 Bearings and Lubricants

3.8.1 Objectives - The objectives of this test are to verify the adequacy of ground testing and to obtain qualitative design data on bearing operating life and standby capabilities in a space environment.

3.8.2 Background - Space vehicles use a multitude of bearing applications in equipment such as gyroscopes, pumps, solar panels, trackers, and antennas. Whenever possible the bearings are sealed within a pressurized enclosure. However, this is not always practical. For example, passive tracking equipment, such as the Gemini horizon sensor, operates best when the signals are not attenuated and distorted by windows in a pressure compartment. Since the sensors are exposed to space, the associated bearings also must withstand this environment and be specifically designed for it. Factors which affect the selection of the bearing and lubricant best suited for a particular application include operating speed, bearing load, pressure and temperature at orbital altitude, and design life.

In the environment of earth, it is possible to lubricate moving parts to reduce friction and wear. Even without specific lubrication, exposed solid surfaces absorb films of oxygen and water vapor which act as lubricants. In the space environment, these surface lubricants are lost through sublimation into space, absorption by the solid, chemical reaction with the solid, sputtering from proton collision, chemical change by radiation and mechanical wear between moving surfaces. Once lubricants are removed, the surfaces remain clean and tend to weld to each other, resulting in high coefficients of friction. Reference (1) summarizes the expected requirements for bearing lubricants in aerospace applications. Table 3.8-1 summarizes the bearing and lubricating techniques planned for a group of scientific satellites.

Much ground testing has been performed using vacuum chambers to simulate the space environment. Useful data has been obtained concerning vacuum effects on

TABLE 3.8–1

SPACE LUBRICATION FOR SCIENTIFIC SATELLITES

SATELLITE PROGRAM	PRESSURE REGION (TORR)	SPEED	LOAD	LUBRICATION APPROACH			
				LOW VAPOR PRESS. OILS & GREASES	LAMINAR SOLIDS	PLASTICS	METALLIC FILMS
EGO-POGO	10^{-12} to 10^{-8}	Low	Mod	G–300 and Apiezon "K" on ball bearings	MoS_2 burnished on ball bearings, Ever-lube coated shaft, MoS_2 run in on level wind and impregnated into gear.		Au plated ball bearings & gear, 80% Ag sleeve bearing, chromium plated gear.
ARENTS	10^{-12}	High	Light	F–50 oil, G–300 grease, Aero Shell 15 Grease, MIL L–6085A oil – all on ball bearings.		Teflon Fluorosint	
OSO	10^{-8}	Low	Light	Apiezon "C" oil	MoS_2 film on ball bearings & chain drive.	Rulon retainers	Coin Silver slip ring & brush.
NUMBUS	10^{-9}	Low	Light	G–300 grease, F–50 oil, Windsor Lube oil MIL L–6085A.	12% MoS_2 in brush.		88% Ag in brush.
TIROS	10^{-8}	Mod	Light	Windsor Lube MIL L–6085A	Graphite	Nylon Bartemp	
OAO*	10^{-9}	Low Low	Light High (10 lbs.)		MoS_2** MoS_2 (Vac Kote)	Teflon Rulon	Ag**, Au

**CBS MoS_2–Ag coating

*One synthetic sapphire pivot bearing used in a stepping motor – low speed light load.

bearings and lubricants. Lockheed performed a series of tests using motors of a standard design (Reference 2). These were fitted with various bearing designs and lubricating schemes and tested at vacuum levels of 10^{-7} to 10^{-9} torr. and temperatures ranging from 160 to 200°F. Other tests have been performed, only on lubricating schemes, using simulated bearing surfaces such as a rider on a rotating disc. Still other tests have been performed on bearings and lubricants by using test fixtures which could vary bearing speed and loads over a wide range.

The environmental simulation accomplished by these tests is not fully adequate. Existing test facilities cannot achieve pressures much below 10^{-9} torr. while the space environment is expected to have a pressure range of 10^{-8} to 10^{-12} torr. In addition, lubricant flow in a zero-g environment cannot be simulated nor is it possible to simulate the entire space environment in one test. An orbital test will provide a realistic environment and provide data for correlation of ground test results.

Some information on friction and wear is obtained from the operation or failure of various spacecraft components. However, such an approach is (1) slow, (2) not quantitative, (3) unreliable, and (4) may mean jeopardizing a scientific experiment or the whole mission. A typical example of such data was the failure, attributed to a bearing, of a Nimbus solar panel. These factors make it desirable to design specific bearing experiments to be flown on available vehicles. Such a test is attractive because the experiment imposes few restrictions on the carrier vehicle. An experiment involving low speed applications was flown on Ranger 1 (Reference 3). Because the vehicle failed to achieve its highly elliptical orbit, the pressure regime was only 10^{-6} to 10^{-9} torr. The total test time was five hours.

An orbital experiment could be designed using any of the three ground testing methods. The Ranger tested lubricants on simulated bearing surfaces. Bearing tests performed on special fixtures,to provide variable speed and load conditions, would provide the most complete test. But since one of the primary aims in bearing tests is to verify ground test results, testing the bearings in a manner similar to that used by Lockheed will provide this data using a relatively simple and reliable experiment.

3.8.3 Functional Description - The multitude of applicable bearing designs and lubricants which are available make it difficult to determine how many and which to test. For this reason, the approach used in this experiment description

BEARINGS AND LUBRICANTS EXPERIMENT
FUNCTIONAL BLOCK DIAGRAM

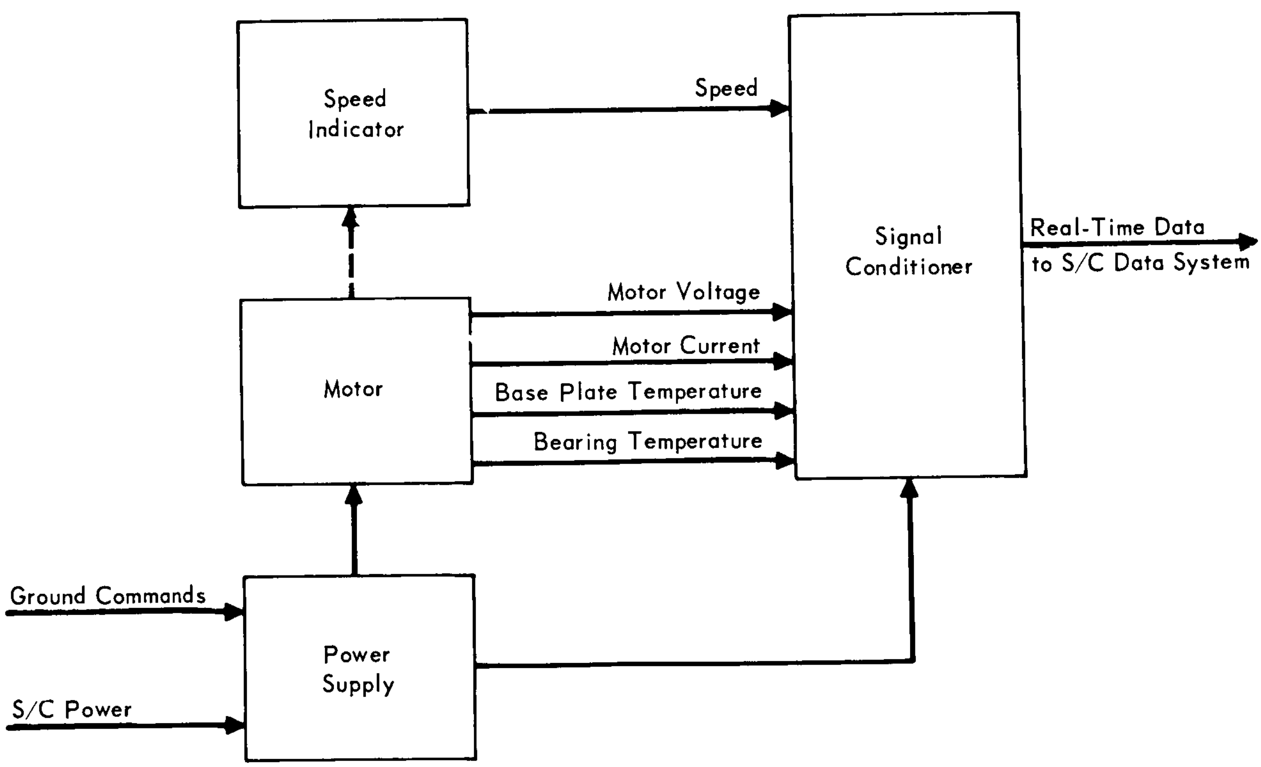


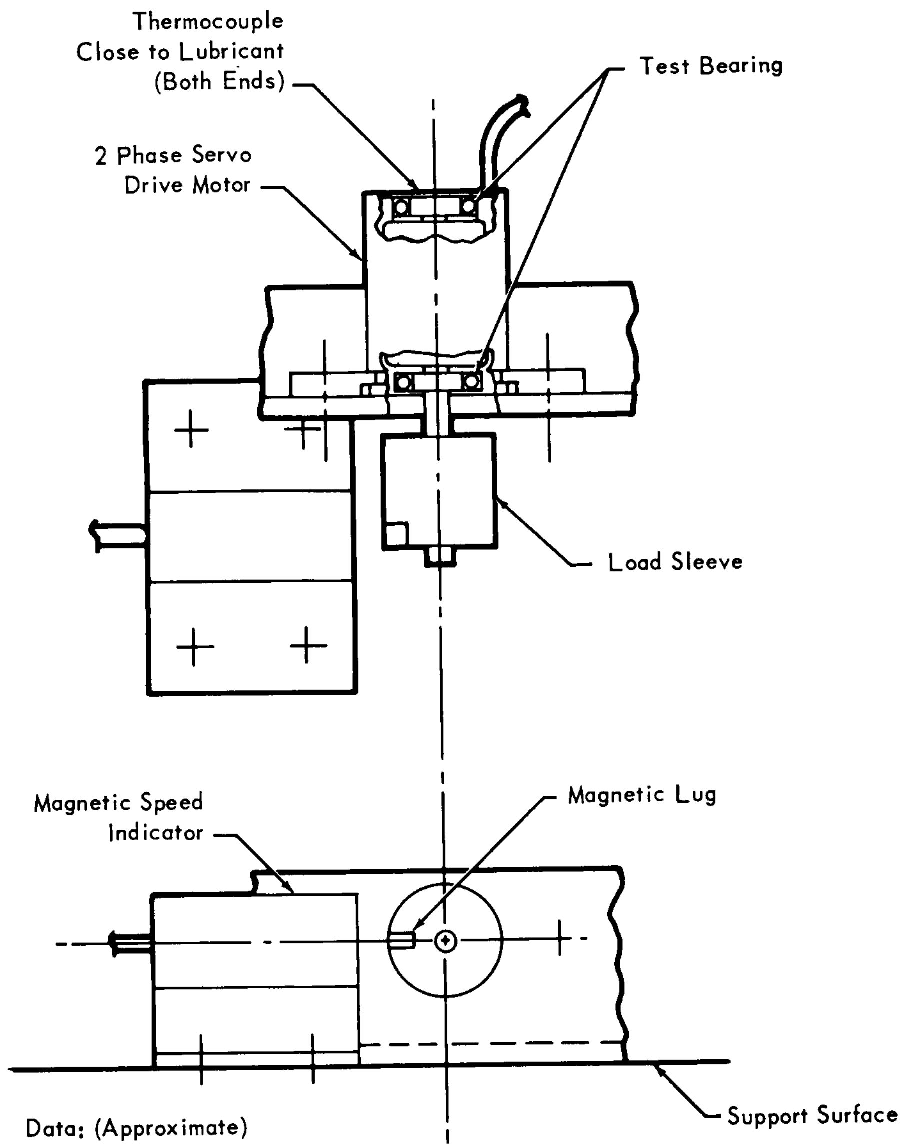
FIGURE 3.8-1

is to define a typical test configuration involving a bearing and lubricant without specifying their exact nature. A functional block diagram for the typical test is shown in Figure 3.8-1. The experimental equipment is the motor and its speed indicator such as the one shown in Figure 3.8-2. The programmer, signal conditioner and power supply comprise the experimental support equipment.

Experiment Equipment - The bearings under test are those which support the rotor of the two-phase servo motor. Based on power restrictions, a motor was chosen which requires approximately 75 milliamps per phase. The motor speed is a function of the input voltage and the bearing friction. By maintaining the voltage constant, the friction can be determined by measuring the speed. The proposed speed indicator is a magnetic device which produces a voltage pulse each time the magnetic lug passes close to the coil. As an alternate approach, friction levels could be measured using strain gauges. While this approach would require a more sophisticated design, it would also relieve the complexities in monitoring the motor speed.

Test Method - The test method required by this experiment is extremely simple. After the carrier vehicle has attained orbit, motor power is applied upon ground command. Data will be monitored once each orbit, upon ground command, for the first four to six orbits. Thereafter the data will be monitored once per day. Operating life will be tested for a period of two weeks and then motor power will be removed. Run-down time will be monitored. A two week standby test will then be conducted. Data will be monitored once per day to provide a time history of both plate and bearing temperatures. This four week test cycle will be repeated until a failure occurs or until the carrier vehicle can no longer support the experiment power requirements. When bearing friction has become sufficiently high to lower

TYPICAL BEARING TEST UNIT



Data: (Approximate)

Weight	.50 lbs.
Volume	8.2 cu.in.

FIGURE 3.8-2

TABLE 3.8-2

BEARINGS AND LUBRICANTS EXPERIMENT PHYSICAL PARAMETERS

EQUIPMENT	SIZE		WEIGHT (LB.)	POWER	
	VOLUME ($FT.^3$)	L-W-H (IN.)		PEAK (WATTS)	NOMINAL (WATTS)
Bearing Package	.006	3 x 3.5 x 1	0.5	3.5*	2.8*
Signal Conditioner	.002	3 x 1.5 x 0.5	0.1	5*	1*
Power Supply	.007	2 x 2 x 2	0.6	13	6
Total	.015		1.2	13	6

*Supplied by Power Supply

motor speed to 90 percent of rated, a daily run-down time test will be conducted. Power will be immediately restored after run-down.

3.8.4 Experiment Physical Parameters - Physical parameters are defined in Table 3.8-2 for the typical test configuration and the experimental support equipment required for the individual test unit. The physical parameters for multiple unit experiments can be estimated by applying the following rules.

1. The size and weight for each additional bearing package will be similar to the present one.
2. The programmer volume and weight will be increased by 10 percent of the present value for each additional unit.
3. The volume, weight, and power of the signal conditioner is directly proportional to the number of experimental units.
4. The volume and weight of the power supply will increase by 15 percent of the present value for each additional unit.

TABLE 3.8-3

BEARINGS AND LUBRICANTS EXPERIMENT DATA PARAMETERS

DATA POINT	SIGNAL FORMAT AND FREQUENCY	PARAMETER RANGE	SAMPLE RATE (/SEC.)	ACCURACY	ESSENTIAL	DESIRABLE	REAL TIME
Motor Speed	Pulse, 0–200 pps	0–12,000 rpm	1	± 50 rpm	√		√
Bearing Temperature	Analog, DC	0–400°F	1	± 15°F	√		√
Base Plate Temperature	Analog, DC	0–250°F	1	± 10°F	√		√
Motor Voltage	Analog, 400 cps	0–30 volts	1	± 2 volts		√	√
Motor Current	Analog, 400 cps	0–200 ma	1	± 10 ma		√	√
Motor Frequency	Analog, 400 cps	380–420 cps	1	± 5 cps		√	√

Note: All data points will be monitored upon ground command..

3.8.5 Data Parameters - The objective of this experiment is to measure bearing life and operating characteristics in an orbital environment. To accomplish this goal, the parameters listed in Table 3.8-3 are monitored.

3.8.6 Vehicle Orbit and Attitude Requirements - The only vehicle constraint for this experiment is that the orbit altitude and eccentricity must be compatible with the desired pressure for the experiment. There are no attitude requirements.

3.8.7 Experiment Support and Data Handling Requirements - The data handling requirements discussed in Appendix B are applicable. The transponder is not required since vehicle orbital position is not required by the experiment. Likewise, since only real time data is to be taken, there are no recorder requirements. The bearing package must be maintained at a temperature of + 80 ± 20°F. There are no special mounting requirements. The experiment will function in conjunction with two ground commands. The "TEST" command will turn power to the motor on and off. The "MONITOR" command will start and stop real time data transmission.

Based on a two week on and two week off schedule, the experiment will require approximately 2,900 watt-hours per month. A unique signal conditioning requirement is discussed below.

Signal Conditioner - Monitoring the pulse signal from the speed detector presents a potential problem to the signal conditioner. A pulse rate detector will provide a DC output signal which is proportional to the pulse rate input with a 5 percent uncertainty. A circuit consisting of counters,shift registers and associated logic can provide an accuracy of 1 percent but at a higher size, weight and power. A compromise can be achieved by using the pulse rate detector, but switching between the output of the speed detector and that of a pulse reference generator. Within its own uncertainty, the generator will supply a calibration signal for processing the detector output.

3.8.8 Experiment Flight Test Plan - The development plan for the program is detailed in Figure 3.8-3. The gating factor in the seven month time base is the three month design and fabrication time required for the bearing package. The flight test sequence utilizes the following steps which take place shortly after orbit insertion.

a. Experiment is turned on by ground command.

b. Data is monitored for one minute upon ground command; once each orbit for the first six orbits, once per day thereafter.

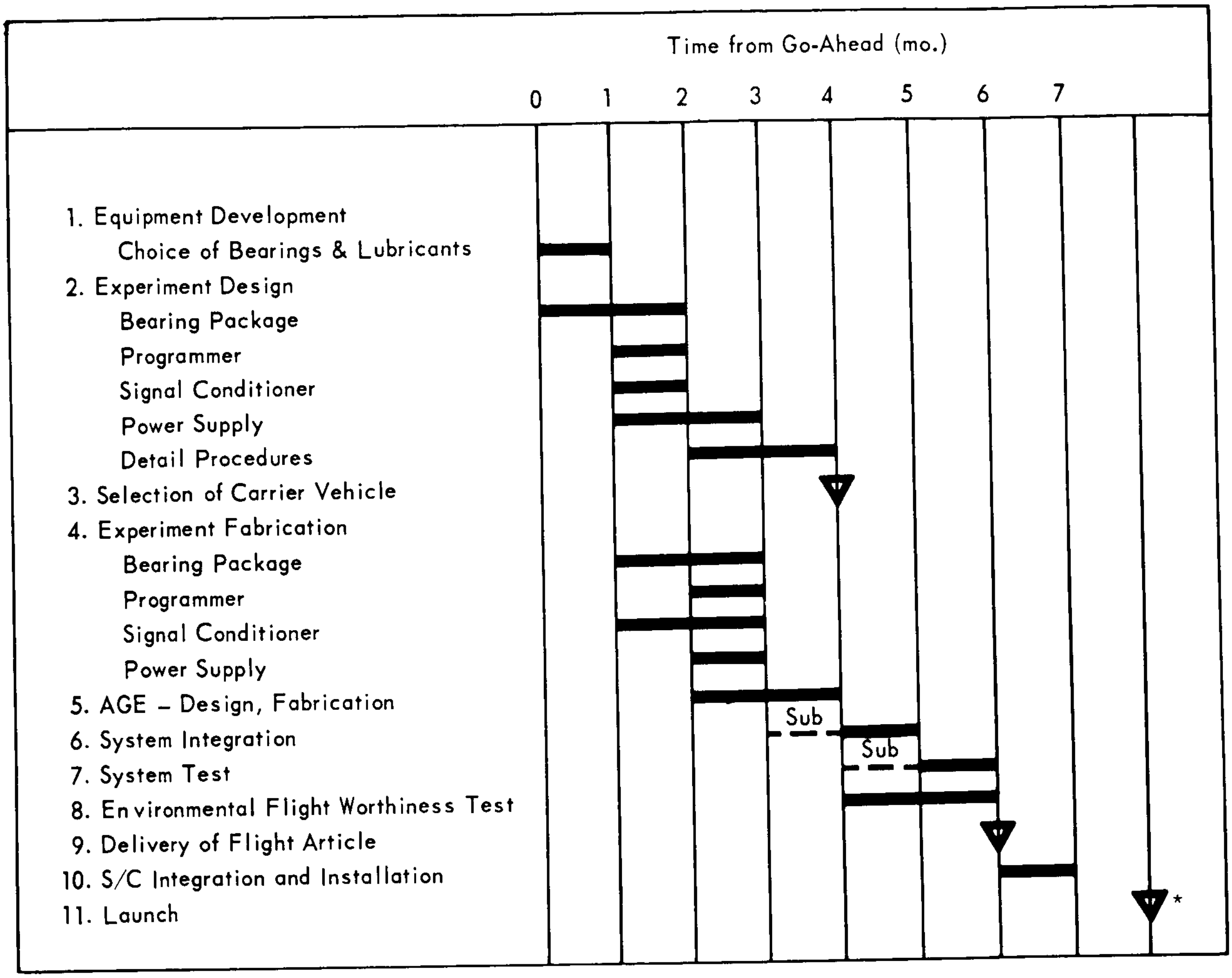
c. After fourteen days the experiment is turned off upon ground command and a run-down test is conducted.

d. After fourteen days of standby, repeat steps (a) through (c).

e. This sequence will be repeated until power is no longer available, or a failure occurs.

Data processing will consist of determining and recording each data point for each monitor period.

BEARINGS AND LUBRICANTS
EXPERIMENT DEVELOPMENT PLAN



*Depends on Carrier Vehicle. May vary from 1 to 8 mo. after installation

REFERENCES

1. General Electric (SSL) Report R64SD38, "Survey of Aerospace Requirements for Bearings and Lubricants", 1964

2. Young, William C., "Lubrication in Vacuum", The Journal of Environmental Sciences, February 1964

3. Rittenhouse, J. B., et al, "Friction Measurements on a Low Earth Satellite", JPL/CIT Report 32-402, 1963

FIGURE 3.8–3

4. CATEGORY C EXPERIMENTS

The Category C experiments cover a broad spectrum of experimental areas. In general, the devices or concepts require further detailed study and analysis. The fifteen experiments are considered worthwhile for further study, but were not fully explored on this program because of development status or vehicle design constraints.

The time scale for the development of some of the equipment is not compatible with this study program; however, the application of these devices appears promising. The experiments requiring further development effort, such as hardware development and target definition, or which have questionable application in near future NASA missions include Automatic Landmark Tracking, Microwave Radiometric Local Vertical Sensor, Cryogenic Gyro, Densitometers, and V/H Sensing.

Several of the experiments place a major constraint on the design, control or mission parameters. Those experiments requiring special carrier vehicle considerations for meaningful experiment definition include Planet-Moon Vertical Sensor, Gravity Gradient Controls-Active Damping, Temperature Rate Flight Control System, Rendezvous Sensor Subsystem, Fluid Systems, Control Logic, Reaction Jets, and Passive Control Techniques.

The Extravehicular Control experiment is a very specialized case and probably requires use of a manned vehicle. The man is not necessarily required to leave the vehicle, but rather, he performs the experiment operations and evaluations. Another experiment which would benefit from being performed in a manned vehicle is the Space Environment Test. The man would perform the measurement evaluation, failure mode analysis and revision of test methods as needed.

For each Category C experiment, the format includes an objective and a discussion of the background and suggested or possible tests. The majority of the experiments require either additional development and ground tests or carrier vehicle selection to define meaningful orbital tests.

4.1 Planet-Moon Vertical Sensor

4.1.1 Objective - The purpose of this experiment is to evaluate the in-flight measuring capability of a planet-moon vertical sensor when operating against actual targets.

4.1.2 Background - Most space vehicles require reference sensors to establish vehicle attitude with respect to a set of coordinate axes. When the vehicle is necr a planet, a horizon sensor is usually used to measure the vehicles attitude with respect to the center of the planet. The most common method of determining local vertical is through use of the planets thermal emission characteristics.

Experiment 2.4 and 2.5 describe in detail the problems associated with simulation of the earth for ground test evaluation of horizon sensor. Similar problems may exist for simulation of other planets with atmospheres, such as Mars and Venus. More measurements of the planets from a lesser distance than from earth will be needed to determine the full extent of the problems associated with simulation. Atmospheric attenuation in narrow spectral bands and beyond fifteen microns as well as the distance to the planets make detailed mapping and measuring of the planet characteristics extremely difficult.

Simulation of the moon does not present the same identical problems as earth simulation. Since there is no known atmosphere on the moon, the gradient slope is infinite. The moon presents problems of temperature extremes and variations in reflectivity which are similar to the atmospheric anomaly problem.

A plot of the apparent temperature distribution along a diametrical scan of the earth, moon, Mars and Venus is shown in Figure 4.1-1 (Reference 1). The figure is a great over-simplification but is useful for comparison of the gross radiation characteristics of the four objects. The dynamic range of maximum to minimum radiation levels may be quite different in restricted spectral bands. Reducing the spectral bands will reduce the apparent radiance but may also reduce the dynamic range.

RADIANT EMISSION FROM PLANETS

(REFERENCE 1)

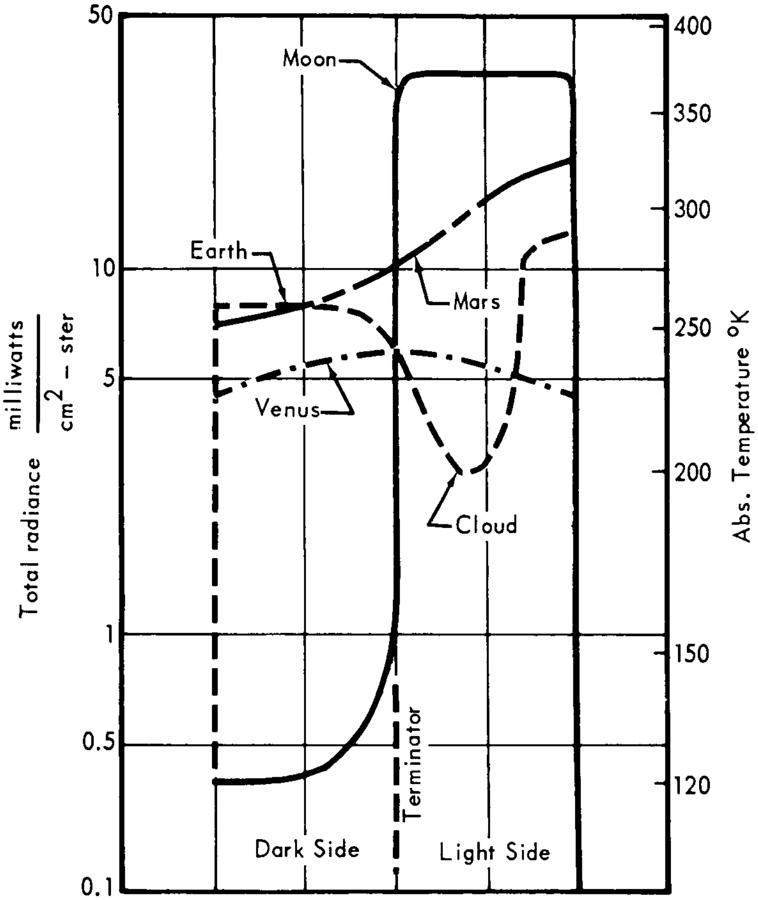


FIGURE 4.1-1

From Figure 4.1-1 it is apparent that the moon presents the greatest problem of dynamic range. The ratio of the radiance of the light side to the dark side of the moon is about 100 to 1, whereas the ratio of the dark side of the moon to space is only about 50 to 1. Sun illuminated crator edges and shadows cause additional "hot" and "cold" areas across the face of the moon which are not illustrated on the figure but have a dynamic range of about 100 to 1.

The large dynamic range of radiance levels presents several problems to sensor designers. The radiant emission is highly non-uniform across the surface. For accuracy, the system must be independent of the radiance level. The detection system must provide usable signals from low energy sources such as the dark side of the moon, and yet not be overly sensitive to direct solar radiation. Several techniques have been developed in an attempt to minimize the problems. The sensing schemes can be shown to be versions of the following (Reference 2):

a. Passive Scan (requires a spin stabilized vehicle) - Used on some TIROS and Explorer vehicles.

b. Conical Scan (Scanning performed by unit rather than vehicle.) - Used on Mercury, Nimbus and other vehicles.

c. Linear Scan (Edge Tracking) - Used on Orbiting Geophysical Observatory, Saturn and other vehicles.

d. Nutating Scan (Edge tracking and azimuth scan.) - Used on Gemini.

e. Illumination Comparison (Radiometric Balance) - Used on Ranger vehicle.

Several companies have performed design analysis, studies and breadboard fabrication of proposed moon-planet vertical sensors. Under a contract with Jet Propulsion Labs, Barnes Engineering Company has fabricated prototype versions of a Lunar-Planet Horizon Sensor (LPHS) (Reference 3). The unit uses a detector array of thermopiles (Reference 1) and electronic linear scanning. Development of this unit is more advanced than other known units.

Ground testing of the LPHS has verified the design criteria but has not provided the confidence desired for space application. The lack of confidence is due to the ground test simulation inadequacies. The unit should be tested in orbit to obtain the desired confidence in the system's capability of measuring vehicle attitude with respect to the moon or planets local vertical.

4.1.3 <u>Test Method</u> - Of the several methods which could be used to test the Lunar-Planet Horizon Sensor in orbit, a test in three phases is recommended.

Phase I will test the design concept, Phase II will determine the accuracy of the unit measuring earth local vertical and Phase III will determine the accuracy of the unit measuring the local vertical of the moon. A discussion of the test method for each phase is contained in the following paragraphs.

Phase I - Test of Design Concept - Two optical heads of the Barnes Engineering Company Lunar-Planet Horizon Sensor (LPHS) are mounted on an earth oriented, earth orbiting vehicle. The units are hard mounted to the vehicle inasmuch as operation in a nulling loop is not required. The electro-optically commutated thermopile detector outputs are transmitted to ground tracking stations for recording and later data reduction. Also telemetered are time, vehicle attitude, sensor temperature and applied voltages.

The test is to determine the ability of (a) the sensor thermopiles to detect the earth-space gradient, (b) the photo-switching circuits providing low noise commutation, and (c) signal processing circuits converting the discretely sampled data to a measure of earth local vertical or gradient position. Considerable care must be exercised in the design of the signal conditioning section of the experiment in order that the discrete sampling of the ninety thermopiles per sensor head does not exceed the standard IRIG telemetry bandwidth.

Phase II - Earth Sensing Accuracy - A complete LPHS is mounted in an earth oriented orbiting vehicle. The LPHS will consist of four sensing heads and the signal processing electronics. The test is conducted in accordance with Experiment 2.5 "Horizon Sensor Accuracy", or 3.4, "High Reliability Horizon Sensors", depending upon the expected performance of the test unit.

Phase III - Moon Sensing Accuracy - A complete LPHS, consisting of four sensing heads and electronics package, is mounted on a vehicle intended for a moon mission. The planned mission may be moon fly-by, circum-lunar or lunar impact/landing. The test unit operates during the lunar approach phases. A master reference system such as a star tracker will be required for this phase. If the LPHS operates properly against the moon (with its large dynamic range), it can be assumed that performance against Mars and Venus will also be satisfactory.

REFERENCES

1. Gerald Falbel and Robert Astheimer, "Infrared Horizon Sensor Techniques for Lunar and Planetary Approaches", presented at AIAA G and C conference, 12-14 August 1963 at MIT, Paper No. 63-358

2. Barbara K. Lunde, "Horizon Sensing for Attitude Determination", presented at Goddard Memorial Symposium of American Astronautical Society, Washington D.C., 16-17 March 1962, Paper No. G-485

3. McLauchlan, J., "IV. Spacecraft Control: A. Advanced Development Horizon Scanner for Lunar and Planetary Applications.", JPL Space Programs Summary No. 37-23.

4.2 Gravity Gradient Controls - Active Damping

4.2.1 Objective - To determine the improvement in alignment accuracy and damping of a passively oriented satellite augmented by an active control system.

4.2.2 Discussion - Certain satellite missions require aligning one axis with the local vertical continuously. For satellites having desirable inertial configurations and where time to damp liberations is not critical, passive damping techniques, as discussed in Paragraph 3.1 are quite satisfactory. However, satellite design limitations or mission requirements may make passive damping techniques impractical. Design limitations may prevent the use of an extensible boom deployed along the earth pointing axis, as it might interfere with cameras or antennas pointed toward earth. The unaugmented inertia configuration with lower gravity gradient torques may under go large liberations requiring active control mechanisms for damping to within a desired alignment accuracy of 2°. The mission may require occasional vehicle attitude changes such as orienting one face toward the moon or other directions away from the earth vertical. This would require active control mechanisms such as reaction jets.

In the past, Discoverer satellites have successfully used cold gas reaction jets to control liberations. The system depends upon attitude information supplied by three orthogonal position gyros. Their signals, combined in conventional manner with rate information from three orthogonally-mounted rate gyros, are used to control the vehicle attitude through the control system. In Discoverer vehicles, when coasting in orbit, the error signals actuate cold gas reaction jets through specially designed proportional gas valves.

Other active control mechanisms for use in damping the liberations of earth local vertical oriented satellites would be candidates for orbital test. These include control moment gyros, magnetic torquing devices, reaction wheels, solar panels, and aerodynamic surfaces. Error signals to actuate these devices could be generated by an earth local vertical sensor compatible with the desired satellite alignment accuracy.

Of the control devices listed, the magnetic torquer, solar panels and aerodynamic surfaces are sensitive to orbital parameters. Use of solar radiation pressure is only applicable in high altitude orbits where gravitational and magnetic forces are small. Orbital eccentricity determines the time spent in the earth's umbra, and thus the energy available for satellite orientation. Aerodynamic surface controls are only effective in the sensible atmosphere (low altitude orbit) and depend on the vehicle parameter, W/C_DA. The magnetic torquing device is only effective at altitudes where the magnetic field of the earth is relatively strong compared with aerodynamic, solar, or gravitational forces. The available field strength depends on orbit inclination and satellite position in orbit, and therefore does not provide a constant or continuous energy source for control.

Inertia wheels and control gyros are insensitive to orbital parameters and appear to offer the potential of precision alignment with the earth local vertical, or any chosen reference direction. Reference (1) describes a control system using two gyros to provide damping and attitude control.

4.2.3 Recommended Tests - Since reaction jets, inertia wheels and control gyros are unaffected by orbital parameters or field magnitudes, orbital test of these devices should be performed to determine that 2^o pointing accuracy can be obtained and that damping to this value will occur within the required time interval. Orbital test would also offer a final environmental proof and life test for the devices.

REFERENCE

1. Lewis, J. A. and Zauac, E. E., "A Two-Gyro, Gravity-Gradient Satellite Attitude Control System", The Bell System Technical Journal, Vol. XIII, No. 6, November 1964.

4.3 Automatic Landmark Tracking

4.3.1 Objective - The purposes of the suggested experiments are to: (1) collect target signature data on selected natural earth features; (2) demonstrate that an automatic passive optical tracker can acquire and track random earth landmarks which lie along the orbit track; and (3) evaluate the accuracy of tracking the landmarks.

Landmark tracking measurements are intended to serve as input data for a near-earth, self-contained orbit determination system. The landmark data may also be used for alignment of an attitude reference system and for image motion compensation in an optical-camera system. Unknown landmark tracking techniques also have applications for lunar (Reference 1) and Voyager missions where orbit determination, attitude reference alignment or image motion compensation is required.

4.3.2 Background - Landmark tracking for orbit determination is attractive because of the potential measurement accuracy and consequent navigation accuracy for low altitude orbits. Landmark tracking can be broadly catagorized by target type into (a) active or cooperative targets and (b) passive or earth feature targets. Microwave, infrared and optical sensors can theoretically be used against either type of target. The preferred sensor technique depends on many factors including: (1) can either active or passive targets be used or must it work with both; (2) is target identification required; and (3) what are the target signal characteristics (including background noise).

If either active or passive targets can be used, active is preferred because both target identification and signal characteristics can be enhanced. In this case, microwave sensors are preferred because of development status. On the other hand if passive targets must be used and target identification is required, either optical map-matching, infrared tracker or microwave radiometer techniques may be preferred depending on target signal characteristics and other considerations. If identification of passive targets is not required, optical techniques are preferred because many targets with adequate signal characteristics are available. The following paragraphs provide additional discussion on the various techniques for active and passive target tracking.

Active Landmark Targets - As defined here, active targets are those where ground aids are used to enhance identification and/or signal characteristics. Active ground aids include beacon transponders, reflectors, lights, and IR heat sources. Active IR heat sources are not attractive compared to the other techniques due to their extremely high power requirements. Sensors which operate against these targets will not be discussed here. Microwave and optical sensors are discussed further.

Microwave Techniques - The best known technique uses a vehicle radar which operates against a transponder or reflector located at a known ground location. The transponder is preferred over the reflector approach because both vehicle transmitted power and antenna size can be reduced. In addition, the signal to background noise ratio can be improved and transponder encoding can be used to enhance identification. One navigation concept, designated AROD and recently funded by MSFC to Motorola for microwave equipment development, uses three ground beacons for a trilateration measurement approach to derive vehicle position and velocity. An alternate navigation concept uses one-at-a-time ground beacons to derive vehicle position and velocity data. For both navigation concepts either range or range rate may be measured; gimbal angle data is usually not employed. Reference (2) proposes a self-contained navigation satellite experiment based on the single beacon, range only measurement concept.

Aircraft have used variations of this approach extensively and it is anticipated that most of the data required to develop an orbital system can also be obtained from high altitude aircraft. In any event, aircraft testing should be used to the fullest extent prior to orbital proof testing.

Optical Techniques - Target types include single light sources to enhance detection and patterns of lights to aid both detection and recognition. The ground support requirements for optical targets are basically the same as for microwave targets, i.e., a sufficient number of targets must be strategically located around the world to provide navigation fixes. Optical sensors have the disadvantage of being susceptable to local weather conditions which may prevent target tracking. The only known use of these techniques has been for pilot aids in landing aircraft.

Passive Landmark Targets - These targets include those where only natural earth phenomena are used, i.e., no intentional active radiation is employed. The sensors may be active such as radar map-matching systems or completely passive such as microwave thermal radiometers, IR tracking systems, or optical map-matching systems. Reference maps for optical and radar map-matching may be generated from previous flight films, or they may be generated in near real-time using storage methods to remember the pattern long enough to use optical or electronic correlation techniques for tracking. Some of the above sensor techniques are not attractive for navigation functions except in special cases. For example, the size, weight and power for radar map-matching systems might preclude the use of this technique unless other mission functions require that the radar be included. Map-matching using reference maps obtained on previous flights is not attractive except in special cases because of the difficulties in obtaining the reference maps and in reducing the maps to a usable format. The following paragraphs provide additional discussion on thermal systems (both IR and microwave) and optical map-matching systems which are based on real-time reference maps.

Thermal Systems - Thermal radiation contrast in either the infrared or microwave regions of the spectrum can be used to identify the landmark. Several natural earth features exhibit sufficient contrast with the surrounding background to be usable. These include volcanos, geysers, hot springs, islands, lakes and oases. Reference (3) presents study results regarding these phenomena and suggests possible instrument designs.

The IR landmark tracking systems will encounter problems common to the IR horizon sensor such as clouds, haze, etc. which can obscure the landmark during part or all of an orbital pass. In addition, the differential temperature between a target and the surrounding medium can vary markedly. For example, the differential temperature between an island and water can even pass through zero making discrimination difficult.

A system utilizing a microwave thermal radiometer as a sensor has the advantage of being less susceptible to obscuring clouds, fog, etc. Also, the system can sense larger thermal differences between landmark and background in some situations where little or no differential is available to the IR sensor.

Microwave thermal radiometers are being employed for land mapping and are being considered for other applications such as orbital true vertical sensing. The main limitations are size, weight, and beam angle. Developmental work is being performed in these areas by several manufacturers including Collins Radio, Nortronics and Sperry.

A considerable amount of basic data and analysis is required prior to effective utilization of either an IR or microwave sensor for passive landmark tracking. Particular problem areas, including the following, must be resolved.

1. The landmark characteristics such as, size, shape, and degree of isolation.
2. Number of landmarks required per orbit based on sensor altitude and probability of freedom from obscurity of the landmark.
3. Effects of weather conditions on attenuation of landmark and generation of false targets.
4. Landmark and background topographical characteristics with respect to aspect angle and viewing direction to determine the length of time available for tracking the landmark.
5. Correlation of the physical landmark centroid with the thermal centroid.

The above problem areas assume that landmark identification is required. If identification is not required by the navigation system, some of the mission dependent problem areas may become less critical, e.g., the number of targets required and the tracking time available. If random unidentified targets can be used, the sensor system can search until <u>any</u> target with sufficient contrast is acquired and tracked. Optical systems appear particularly attractive for use against random targets since it is known that a sufficient number of usable targets are available.

<u>Optical Systems</u> - Optical tracking systems based on electronic area correlation have been proposed for tracking unknown landmarks. Detailed target characteristics are not critical and many targets with sufficient contrast are available. On the other hand, local weather conditions may obscure the targets and the tracker may track cloud tops unless a discrimination method is designed into the system.

4.3.3 <u>Suggested Tests</u> - Two experimental areas are suggested as candidates for orbital test. In the first, IR radiometers are flown on a manned, earth-oriented vehicle to obtain data on specific target signature characteristics. In the second area, a passive, optical or IR tracker designed to track unknown landmarks along the orbit track is flown and manned or unmanned, earth-oriented vehicle.

The IR radiometer experiment is conducted on a manned vehicle so that man can aid in identifying, acquiring and tracking targets that are thought to have the desired signal characteristics. The radiometer should have a relatively small field-of-view to collect data on selected targets. The data is used to catalog navigation landmarks and to design landmark trackers. An alternate and more complex radiometer test method would use an unmanned vehicle with a navigation computer and attitude reference replacing the man. The navigation computer accepts initial condition data (position, velocity and time) from a ground tracking network and computes radiometer pointing commands for tracking and collecting target data. It may be feasible to combine this radiometer test with the suggested radiometer test in Paragraph 4.8.

One approach for conducting an orbital test of the optical tracker would be to design the tracker as an integral part of a stabilized platform and digital computer. The platform provides a stable reference for pointing the tracker. The computer provides nominal pointing commands and correlation data processing for maintaining track. This approach is attractive in that both in-plane and out-of-plane targets can be tracked. Also, tracking accuracy can be evaluated with the on-board navigation computer. However, the test is quite complex and the alternate approach discussed in the following paragraphs is recommended at least for an initial test. Basically, the alternate approach is to conduct tests with a self-contained tracker. The vehicle and a single tracker gimbal serve as the stable reference. Only in-plane targets are tracked. Correlation for tracking is accomplished by the tracker electronics. Tracking accuracy evaluation is accomplished through use of ground tracking and computing networks.

The optical or IR tracker flight test is based on using a modified V/H sensor (Reference (4), also see Paragraph 4.10). The system is designed to provide an image motion compensation signal to an optical-camera system. The V/H computation is made by tracking an area of terrain as viewed by a vidicon camera and using an electronic area correlation technique. Tracking is accomplished by using a translating lens. Transducers measure the rate-of-change of the lens position to provide the V/H measurement. By using a position transducer to indicate lens position and using simple computations, the angular coordinates of the tracked landmark relative to a vehicle attitude reference could be indicated. Rate-of-change measurement of lateral image drift during the correlation operation supplies a signal to a drift angle servo. This servo rotates the vidicon camera and its optics within a gyro-stabilized gimbal and drives a synchro that supplies a drift angle indication. The V/H sensor performance requirements are to measure V/H to within an 0.5 percent tolerance and the measure the drift angle to an error of less than 15 minutes of arc. This performance was demonstrated in the laboratory (Reference 4) for simulated conditions which included orbital V/H and the position tracking accuracy requirement of 15 arc seconds.

The proposed test method for the modified V/H sensor would mount the sensor to view down along the vertical in an earth-oriented vehicle. The vehicle is also controlled in yaw to the orbit plane. The sensor translating lens can direct the pitch line-of-sight $\pm 20^{\circ}$ in the orbital plane. This freedom permits acquisition of random landmarks that lie approximately 20° forward from the local vertical and track through the local vertical to approximately 20° aft of the local vertical. For a 150 n.m. orbit, average tracking time for each target is about 10 seconds.

This test could be conducted while the vehicle is in contact with a ground station to simplify the experiment programmer and data handling functions. Conduct of the experiment while in ground contact also permits assessment of tracker performance in the presence of cloud cover. Objectives of the test are to determine if the tracker will lock-on and to obtain an estimate of tracking accuracy. Tracking accuracy can be evaluated using the ground network for orbit determination in conjunction with the on-board vehicle attitude reference. To evaluate performance, flight data parameters include the position and rate of the translating lens plus the synchro output from the drift angle servo. These parameters and others would also indicate if the tracker maintained "locked-on". According to Reference (4), the present laboratory V/H sensor experimental model weighs 103 lbs., requires 6430 cubic inches and uses about 300 watts of 400 cycle AC power. An advanced experimental model (V/H only, i.e., not designed for landmark tracking) was estimated to weigh 20 lbs., require 575 cubic inches and use 100 watts of 400 cycle AC power.

REFERENCES

1. DeNazze, E. J., and Dittrich, M. S., "Orbit and Landmark Determination During Lunar Orbit", MIT/T-342, June, 1963.

2. Hughes Aircraft Company Experiment, "Self-Contained Navigation Satellite System Experiment" (5 pages), 16 October 1964.

3. Northrop Nortronics Report, "Proposal for Self-Contained Passive Orbital Navigation System (SPONS)", NORT 64-230, July, 1964.

4. USAF RTD-TDR-63-4061, "Passive V/H Sensor", October, 1963, Prepared by Goodyear Aerospace Corporation, Akron, Ohio.

4.4 Microwave Radiometric Local Vertical Sensor

4.4.1 Objective - Orbital testing of microwave radiometric local vertical sensor techniques would be performed primarily to establish the feasibility of the concept and to obtain data to assist in the development of the device. Ultimately, the accuracy of the device must be determined.

4.4.2 Background - With suitable receivers, the microwave radiation emitted by all materials above absolute zero can be detected and the measurements used to locate the angular direction of the radiating object with respect to the receiver. Microwave radiometric techniques have not been used as extensively as infrared techniques for detection and measurement; however, there appear to be some advantages to the microwave techniques for some practical applications. For example, in detecting radiation from a surface, such as earth or the lunar surface, the microwave system would be less affected by surface irregularities and atmospheric anomalies than an infrared system and thereby give more accurate readings.

The amount of microwave power radiated by a material depends on its temperature and emissivity. The radiometric temperature of a source, as seen by a remotely located antenna, will be modified by the temperature and losses of the intervening material. For the atmosphere, losses in the microwave region are due largely to water vapor and oxygen molecules. Curves in Figure 4.4-1 show typical attenuation for the atmosphere as a function of frequency. Peaks at 23.5 Gc and 60 Gc are due respectively to the water vapor and oxygen molecular resonances.

The radiometric vertical sensing technique is based on the determination of a vertical line around which the atmospheric temperature appears constant. As the antenna look angle is increased in elevation fewer oxygen molecules are encountered by the beam and the antenna temperature decreases. Assuming that the oxygen

TYPICAL ATMOSPHERIC ATTENUATION AS A FUNCTION OF WAVELENGTH

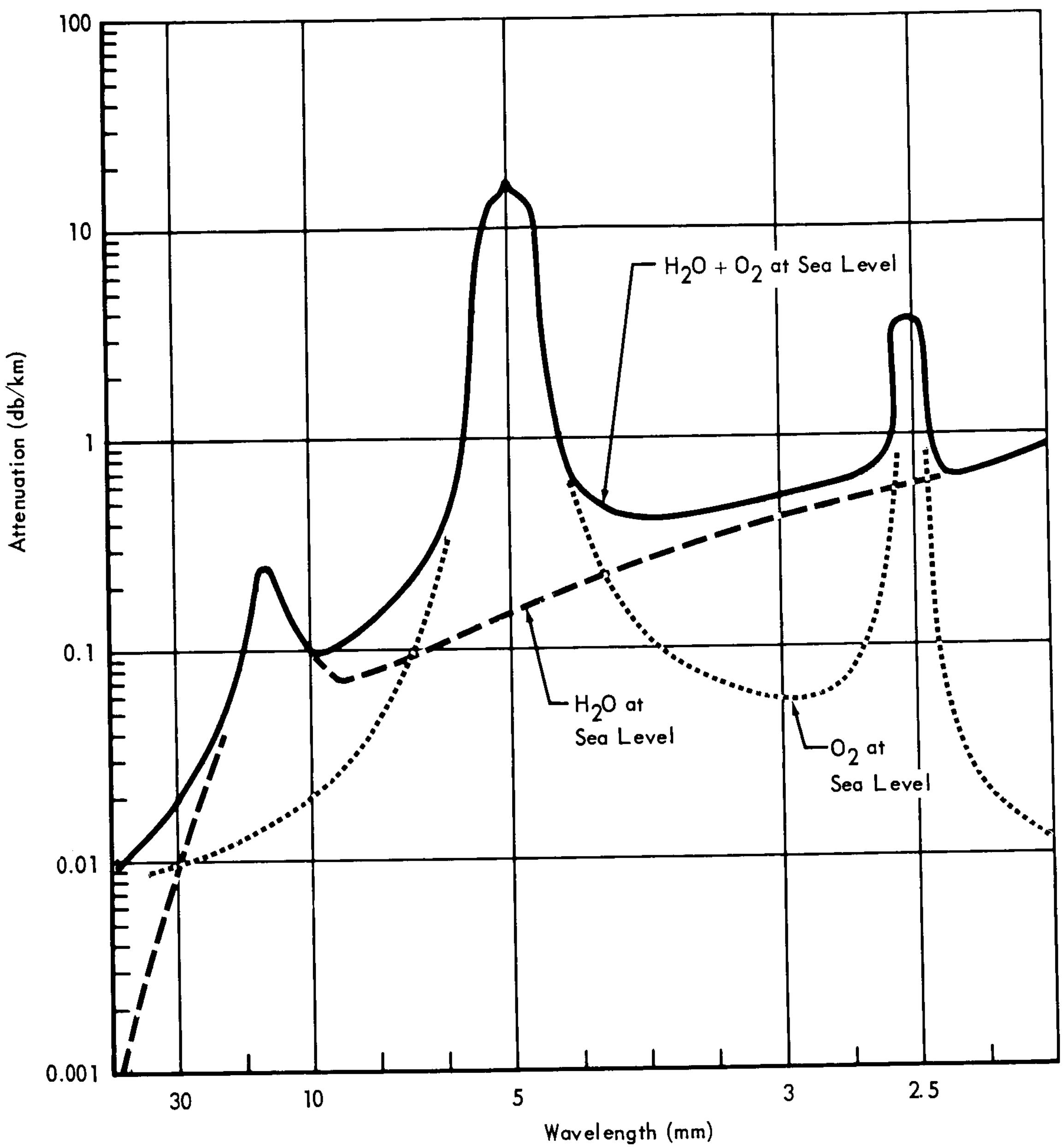


FIGURE 4.4-1

is spherically stratified, a scanning antenna can determine the apparent direction of the earth center to an accuracy of a few minutes of arc.

The accuracy of the system will be affected to some extent by weather and other anomalies but to what degree is now known. Therefore, orbital testing could provide the data to determine the feasibility of the microwave radiometric horizon sensor. The microwave radiometric technique could be used with the earth, moon, sun, or another planet as the target by suitable changes in RF frequency.

Microwave vertical sensors are not off-the-shelf items at this time. The most complete equipment available appears to be that under evaluation in medium altitude aircraft tests by Collins Radio Company. Collins has been working on the oxygen-line radiometry and radiometric vertical sensing techniques under contract of FDL (RTD) Wright Patterson AFB, Ohio.

4.4.3 Suggested Tests - Two areas for possible testing are discussed. In the first area, scientific and qualitative accuracy data is obtained to determine the feasibility of the concept. The second test area is more complex and has a primary goal of obtaining quantitative accuracy data. Both tests are conducted on an earth-oriented vehicle to minimize radiometer pointing problems.

A relatively simple test to determine the feasibility of the concept would be to orbit radiometer equipment which views only a limb of the earth. Infrared horizon sensors are used to orient one axis of the vehicle toward the earth - vehicle yaw alignment is not required. The radiometer collector is gimballed only in elevation (single axis). It may be possible to hard-mount the radiometer to the vehicle such that the collector field-of-view includes the earth limb when the vehicle is earth-oriented. However, the non-gimballed radiometer is very dependent upon vehicle attitude control operation and altitude. The IR horizon sensor provides master attitude reference measurements for estimating microwave radiometer accuracy.

An accuracy test of the microwave system would be conducted in a manner similar to the Horizon Sensor Accuracy test in Paragraph 2.5. This accuracy test is considerably more complex than the first test discussed above and normally would not be conducted unless aircraft tests or these earlier orbital tests have conclusively proven the concept to have potential. Basically the test method is the same as described in Paragraph 2.5. Since a star tracker serves as the master measurement reference, vehicle yaw control is required. The microwave system is designed to track over a wide region of the earth horizon - either by including multiple collectors of the type discussed in the previous paragraph or by rotating a single collector over a sector of the horizon. With either approach, the weight, volume and power requirements for the microwave system are considerable, assuming state-of-the-art antenna techniques.

4.5 Cryogenic Gyro

4.5.1 Objective - The primary purpose of conducting an orbital test of the cryogenic gyro is to demonstrate the potential high accuracy of the instrument in the space environment. Also, the experiments should provide data on operational problems such as maintenance of the cyrogenic temperatures.

4.5.2 Background - The cryogenic gyro holds promise for improving guidance accuracy considerably by the reduction of gyro drift rate. In many respects the gyro is quite similar to the Electrostatic Gyro (ESG) described in Paragraph 2.1. In conventional gyros unpredictable torques resulting from mechanical support friction constitute a major source of drift. The cryogenic gyro, like the ESG, eliminates the friction inherent in mechanical supports by using suspension forces from an applied external field to support a hollow spinning sphere. However, unlike the ESG which uses electrostatic fields, the cryogenic gyro uses magnetic fields for suspension of a superconducting sphere. Since superconductivity occurs at near absolute zero temperatures, the maintenance of these temperatures poses the major constraint on using cryogenic gyros. Despite this constraint, the gyro has several advantages (see Reference 1) including:

a. Superconductors are repelled rather than attracted by magnetic fields; thus the support is inherently stable, unlike electrostatic suspensions which require feedback systems.

b. The current required for support can be trapped in a superconducting circuit and, once suspended, no support power is required.

c. Dimensional instability is virtually eliminated at cryogenic temperatures because material expansion coefficients approach zero.

Various groups have been active in cryogenic gyro research since 1959. JPL has reported some of their results in References 1 and 2. Early work at General Electric is reported in Reference 3. According to Reference 1, the purpose of

JPL's research is to evaluate the known sources of torque which include the non-sphericity of the rotor and the trapped flux in the rotor. These torque sources lead to gyro drifts. G.E.'s work for NASA, known as Project SPIN, is sponsored by Marshall Space Flight Center for application in a gimballed inertial system. In correspondence to McDonnell, G.E. (Pittsfield) concluded that orbital testing of the Project SPIN gyro in its present configuration is neither practical nor particularly useful in terms of obtaining useful data. It was indicated that this conclusion was based on the fact that many of the components of an all cryogenic inertial guidance system are not yet under development. Hence, it appeared premature to attempt to plan or schedule a test at this time.

4.5.3 Possible Orbital Experiments - Because of the constraints imposed by cryogenic temperatures, it is felt that the main use of the gyro will be as a part of an all cryogenic guidance or navigation system. The gyro might also be used in a long term space navigation sextant which incorporated cryogenically cooled detectors for optical instruments such as horizon sensors or star trackers. Since the support and construction of the cryogenic electromagnetic gyro is similar to the ESG, the accuracy constraints for each gyro are imposed by the same type of support circuit and the same manufacturing limitations. That is, the non-sphericity of the spinning rotor coupled with high support forces cause unwanted torques and gyro drift. Also, trapped flux in the gyro rotor and interactions with stray fields cause drifts. For near zero-g space applications, the ESG support levels can be reduced to a very low level (see Paragraph 2.1) thus reducing the drifts caused by rotor non-sphericity. It appears that a useful cryogenic gyro orbital experiment would be to reduce the support forces and monitor the gyro drift rate. Such an experiment might best use a strapped cryogenic gyro and a star tracker master reference in a manner similar to the ESG experiment.

The drift rate experiment, or one which merely performs remote suspension and spin-up, appears to be the simplest orbital test to obtain some useful data pertaining to the cryogenic gyro. The complete inertial guidance system test or a space sextant test requires much additional development before hardware is available.

REFERENCES

1. Harding, J.T., and Tuffias, R.H., "Cryogenic Gyros Levitated by Magnetic Repulsion," Space/Aeronautics, pp 133-141, Sept. 1961.
2. Harding, J.T., "Drift Data for the Cryogenic Gyro," Presented at the 1964 Cryogenic Engineering Conference, University of Pennsylvania Paper, B-7.
3. LaFond, C.D., "G.E.'s New Cryogenic Gyros near Testing," Missiles and Rockets, pp 34-36, 24 July 1961.

4.6 Temperature Rate Flight Control System

4.6.1 Objective - The objective of the experiment will be to evaluate a flight control system using temperature and temperature rate information to control a high L/D re-entry vehicle during re-entry and transition to equilibrium glide.

4.6.2 Discussion - Hypersonic glide vehicles are subject to severe thermal extremes during atmospheric re-entry, equilibrium glide acquisition (transition), and during the equilibrium glide phase. During re-entry, stagnation point heating is critical and severely restricts the re-entry flight path angle to a narrow range of values in order to prevent catastrophic structural failure. For a lifting vehicle the maximum stagnation temperature occurs at the bottom of the re-entry dip, i.e., the point at which the summation of forces in the vertical direction are most unbalanced. Transition from the dip point to the desired L/D equilibrium descent path requires angle-of-attack and/or bank angle modulation, which would result in destructively high underside temperatures if not properly regulated. The same is true of energy management during equilibrium glide. Vehicle maneuvering or external disturbances could cause underside temperatures to exceed safety limits. Since vehicle heating is so critical, it is desirable to directly measure heating rate and temperatures at sensitive areas on the spacecraft and to use this information directly for control or as an override or warning signal in the flight control system.

AC Sparkplug has done simulation work with a manual temperature rate flight control system (TRFCS). The thermal sensor developed for this purpose was flown on an ASSET re-entry flight in December 1964. The purpose of the test was to evaluate the time lags and operational characteristics of the sensors in an actual flight. The data has not been completely evaluated at the time of release of this report.

4.6.3 Recommended Tests - Flight test is recommended to evaluate the capability of a temperature control system to control the atmospheric re-entry of a hypersonic glide vehicle. The test should be designed to prove the concept and to determine if such a system has application for vehicle control or should be considered only to provide warning signals. A vehicle with a lift-to-drag ratio in the vicinity of 1.5 is desirable for this test. The test could logically be performed during flight test evaluation of the M-2 or HL-10 aerodynamic configuration, or on other ASSET flight vehicles.

The experimental system for this test consists of thermal sensors, computer, coupler, autopilot, and actuators for control surfaces and reaction controls. The heating sensors are thermocouples imbedded in the vehicle skin at critical points. Temperature information is processed by a computer which interprets the thermocouple data and converts it to a control signal to the coupler. The coupler converts computer output signals into a form acceptable by the autopilot to use for flight control. Since it is likely that reaction controls will be required to supplement the angular acceleration available from control surface deflection, the autopilot must feed control signals to both actuators.

4.7 Densitometers

4.7.1 Objective - The objective of this experiment is to determine if a densitometer can provide suitable air data measurements for control of lifting hypersonic vehicles during re-entry and equilibrium glide descent.

4.7.2 Discussion - A re-entering hypersonic glide vehicle will undergo phugoid oscillations causing excessive load factor and thermal stress, unless the lift vector is properly controlled. For such control to be exercised, environmental parameters must be obtained from on board air data measurements and used within the vehicle autopilot and control system to provide trajectory damping. For a manned vehicle the environmental data can be displayed to the pilot for manual control or for monitoring the performance of an automatic system.

For craft entering the atmosphere at hypersonic velocities, "Conventional" methods of obtaining air data measurements cannot be used. Probes, pitot tubes, nose booms, etc., used on modern day aircraft would be destroyed in the thermal re-entry environment, or would not function properly in the flow field. A probe would have to extend out past the plasma sheath (in some cases as much as 3 feet) to measure the properties of the atmosphere. Litton Industries and Giannini Controls have developed devices that will provide air data measurements in re-entry. Litton has developed a laser densitometer, while Giannini employs radioactive isotopes in one concept and X-rays in another.

Description of Densitometers - There are three types of densitometers which are candidates for test. They are:

a. Radioisotope densitometer

b. X-ray densitometer

c. Laser densitometer

All devices measure density by comparing backscattered radiation to the source radiation using either the Rayleigh or Rutherford scattering laws to determine the number of molecules in a given volume of atmosphere. Other air data parameters which can be derived from densitometer measurements are: velocity (Mach number), heating and heating rate, angle-of-attack, sideslip angle, and density altitude.

The Laser Densitometer for a Glide Re-entry Vehicle - In this system the received signal photons will be backscattered from air molecules which are at least one meter away from the vehicle. This will ensure that even at the lowest altitudes, the density readings will be outside of the shock front and are therefore measurements of free stream density.

The densitometer consists of a transmitter, an optical window (sapphire or quartz lens), collimating optics and lens stop, a narrow band optical filter, a photomultiplier detector, and all necessary electronics for power supply and readout (see Figure 4.7-1). The laser transmitter, receiving optics, and photodetector are less than 1/4 cubic foot in volume and can easily be mounted within a vehicle nose cone. The power supply and readout electronics will be approximately 1/3 cubic foot in volume and should be packaged separately.

The transmitter system consists of a giant-pulse laser. The general technique of quickly introducing a larger amount of positive feedback into a stable amplifier to produce intense, short bursts of radiation can be applied to laser operation to give the required high peak power. This process is called Q switching and in this case will be accomplished by using a saturable filter. For this purpose, a glass filter will act as a switch, by allowing stimulated emission in the ruby cavity only when the filter becomes transparent, which it does after small amounts of stimulated emission strike it. With state of the art saturable filters, peak powers of ten megawatts are easily attainable.

LASER DENSITOMETER PROBE

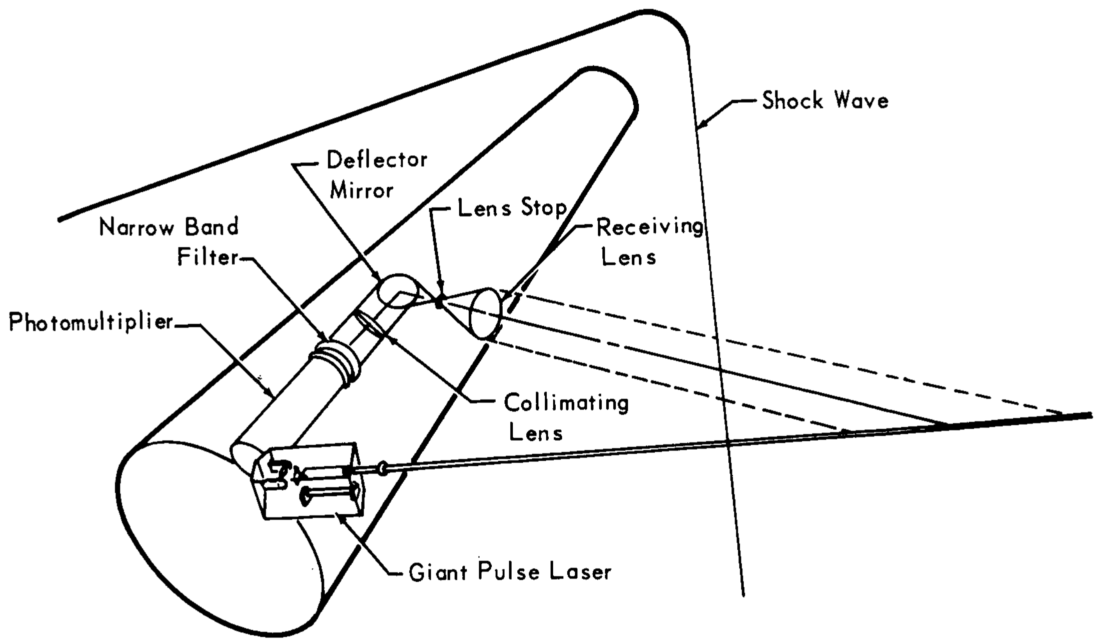


FIGURE 4.7-1

A three-inch-long ruby rod with a sapphire overlay will be used and housed with a xenon flashlamp in a small elliptical cylinder. The energy for the system will be stored by 250 microfarads of capacitance at 2000 volts for a total energy of 500 joules.

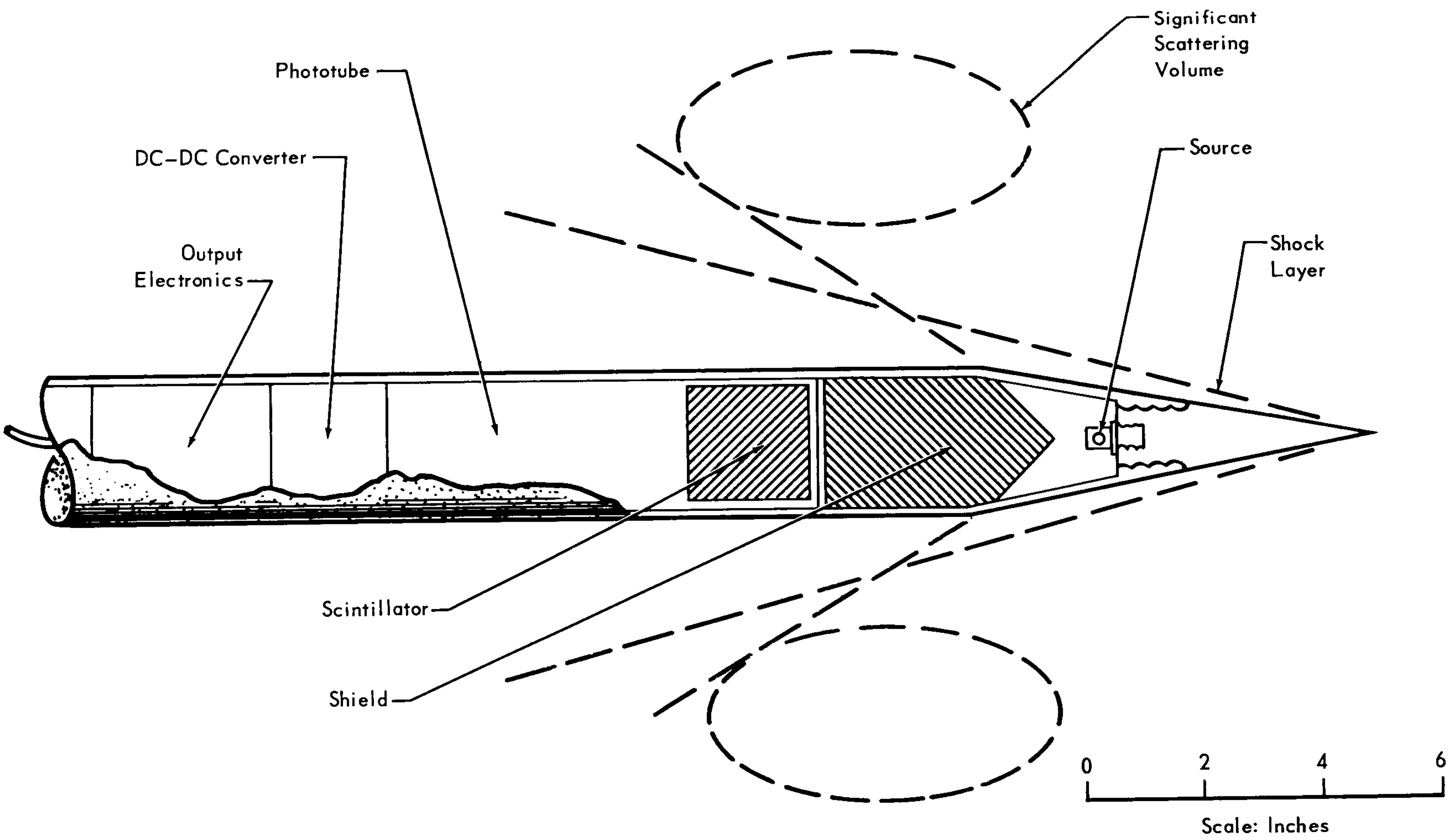
The receiver system will consist of a receiving lens that will gather and focus the backscattered light on the cathode of the photomultiplier which will be gated on during the interval when the laser pulse is present. The photomultiplier output will then be amplified and fed into an integrating amplifer which will give a dc output proportional to the total energy received from the backscattered pulse. A photodiode will monitor the output power level of the initial laser pulse and will also be fed through an integrating amplifier that will give a dc output proportional to the total output energy of the laser. The output of the photomultiplier integrating amplifier divided by the output of the laser monitoring amplifier is proportional to density. The output from this divider circuit provides a high-level dc voltage, suitable for meter display or telemetry.

Because of the large dynamic range of density variation (10^6), it would be desirable to precisely step-reduce the gain of the receiver system so as to limit the dynamic range of the readout equipment. This can easily be done by either precisely stepping the voltage on the photomultiplier dynodes or by gain-stepping the amplifier on the output of the photomultiplier.

Two sources of error may be introduced into the readings of the laser densitometer. The first is associated with the electronic and readout portion of the system. This error should be no more than about one percent of the reading. The second error source will originate from accidental interception of high background radiation sources at the highest altitudes. This error can be avoided by propitious placement of the gauge within the vehicle. Even with severe ship maneuvering, which might subject the gauge intermittently to high background sources, errors of less than five percent will be introduced. Under normal flight conditions, however, errors from background will be much less than one percent.

Radioisotope Densitometer - The optimum configuration for a high altitude air density measurement using a radioisotope technique, disregarding limitations imposed by a particular vehicle, is shown in Figure 4.7-2 In this configuration the distance from source to detector is a minimum consistent with the shielding required to attenuate direct transmission and the separation required to obtain a free stream density measurement. This configuration maximizes air scattering return. The shield and detector are arranged so that any gammas scattered off of the shield or structure cannot reach the detector without first passing through the shield or being scattered by air. The shield weight is minimized, since it is

γ – RAY AIR DENSITY



MEASUREMENT SYSTEM

FIGURE 4.7–2

only imposed between the source and detector. About 0.4 steradians of the gamma emission from the source is utilized for the scattering measurement. Finally, the significant scattering volume is beyond the shock layer, and a large portion of the scattering which does occur within the shock layer is shielded from the detector. The accuracy of the density measurement performed by this configuration as a function of altitude is shown as a function of source strength in Figure 4.7-3.

X-Ray Densitometer - This device is currently under development at Giannini Controls. No detailed information is available at this time. It is similar in operation to the other devices in that it utilizes backscattered radiation to measure free stream air density from re-entry altitudes to sea level. The principle of operation is shown in Figure 4.7-4. The essential elements of the system are: (1) an x-ray source consisting of a power supply and x-ray tube package capable of delivering 70 Kv at 4 ma., (2) a lead collimator, and (3) a scintillator plus photomultiplier detector.

Spacecraft Characteristics - This experiment requires a lifting hypersonic re-entry vehicle to provide the environment necessary for testing the densitometers. An L/D of 1.0 or greater is desirable to insure a shallow glide angle (low rate of descent) necessary to permit a large enough sampling interval for the desired accuracy. The X-15 and ASSET vehicles would be satisfactory for the test. Re-entry altitudes can vary from 200,000 feet to 400,000 feet depending on the vehicle capability. The trajectory limitations depend only on the vehicle heating, load factor, and stability constraints. The primary control requirement is that excessive trajectory (phugoid) oscillations be avoided for this test. It is desirable to test the operation of these devices in a smooth re-entry to

GAMMA BACKSCATTER DENSITY MEASUREMENT ACCURACY AS A FUNCTION OF ALTITUDE FOR THREE SOURCE STRENGTHS

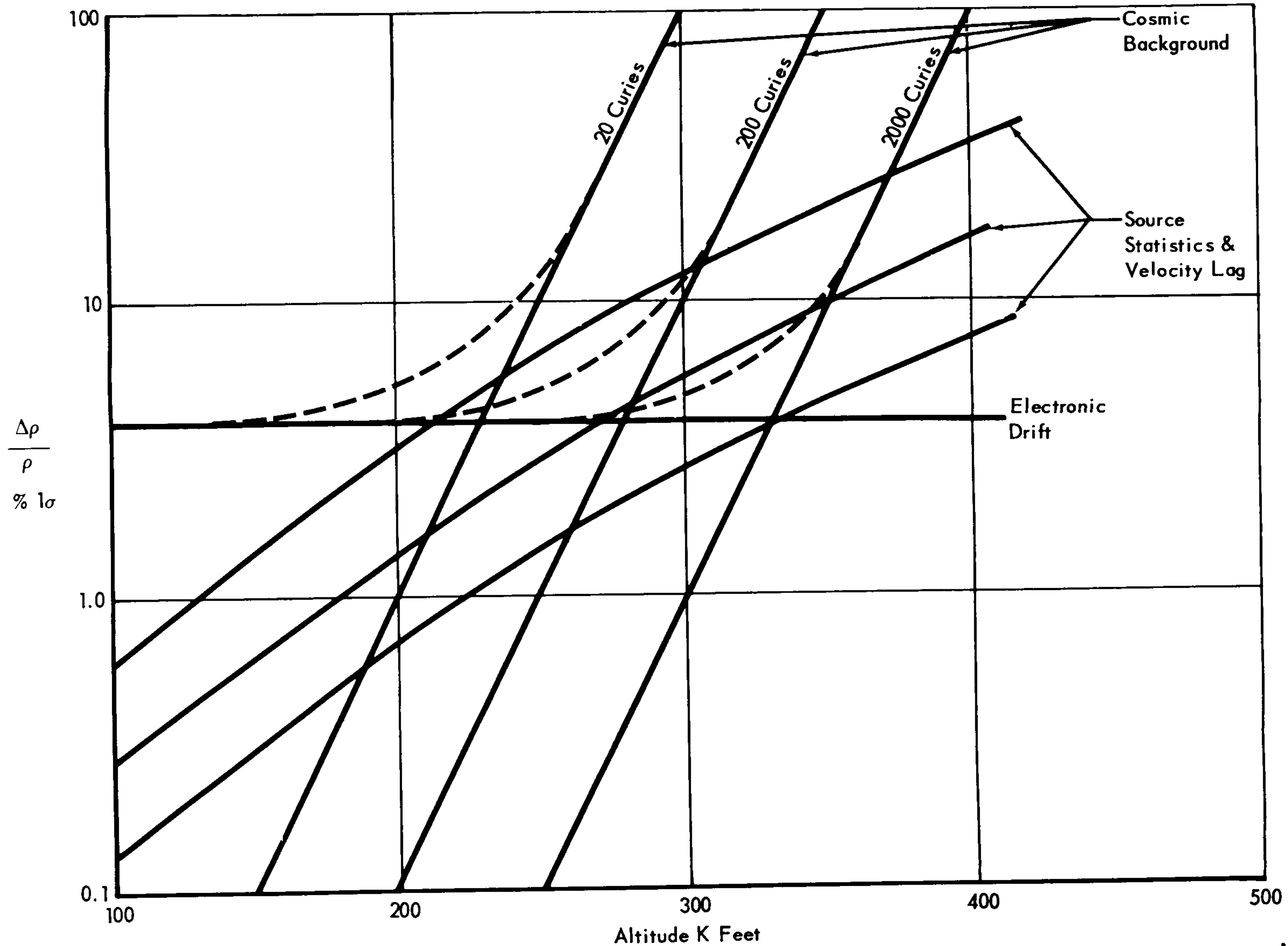


FIGURE 4.7–3

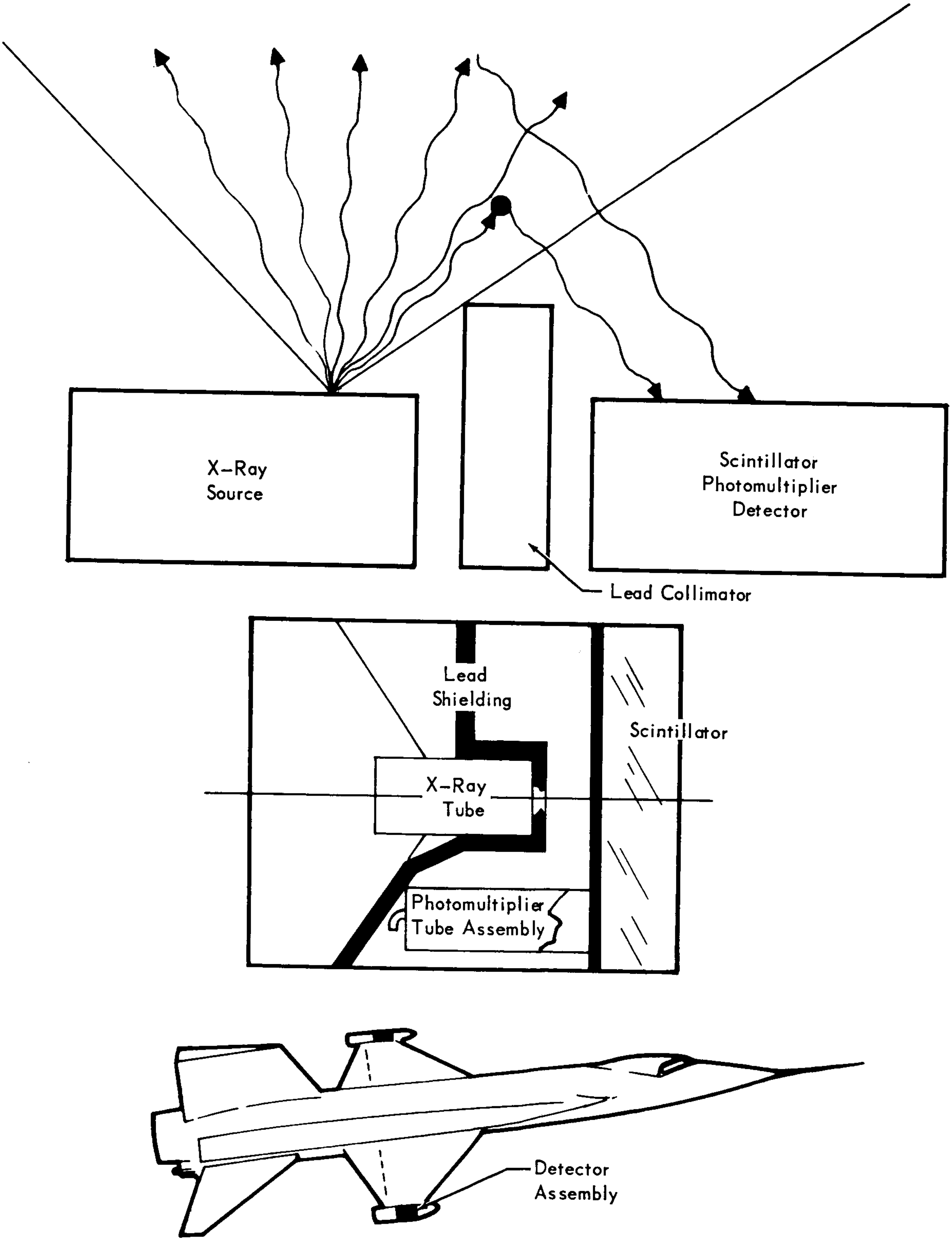

X–Ray
Source
Scintillator
Photomultiplier
Detector
Lead Collimator
Lead
Shielding
Scintillator
X–Ray
Tube
Photomultiplier
Tube Assembly
Detector
Assembly

FIGURE 4.7–4

demonstrate that the physical principle has merit. Oscillating trajectories and vehicles with ablative external surfaces should be considered for later tests.

4.7.3 Recommended Test - The experiment would consist of evaluating the three candidate densitometers on an X-15 or ASSET type re-entry vehicle. The vehicle should have a hypersonic L/D or 1.0 or greater and re-enter at sub-orbital or orbital velocity for this test. Densitometer measurements will be recorded and stored until the re-entry plasma effects have subsided, and then telemetered to ground. Simultaneous measurements of onboard data from the inertial sensors, thermocouples, altimeter, etc., and data obtained from ground tracking will be used to evaluate densitometer accuracy.

4.8 Rendezvous Sensors

4.8.1 Objective - The purpose is to obtain data on the propagation characteristics, the effects of background noise, and the tracking performance of various rendezvous sensor subsystems. The data obtained will be used to improve existing designs and to select appropriate sensor systems for use in missions which require rendezvous.

4.8.2 Background - Sensing techniques applicable to rendezvous include microwave, infrared (IR) and optical devices which may be used to sense or track either cooperative or uncooperative targets. The Gemini and Apollo flights, which require rendezvous, plan to use microwave systems with cooperative targets. Various passive optical devices are planned for use during docking operations. Beacon transponders and visible lights are located in the target vehicles.

Other advanced optical and IR techniques have been proposed for use at long ranges. Laser ranging systems to obtain precise range data may be used with either cooperative or uncooperative targets. Initial laser acquisition and track may be aided by angle-only microwave or passive IR systems. If the IR system is sufficiently precise in angle tracking, it may serve as the primary rendezvous navigation sensor.

Three general areas of orbital testing are of interest for rendezvous subsystems. One area of principal concern is with the effects of background noise. Natural space radiation and both natural and man-made earth radiation are of concern. Typical tests would consist of orbiting narrow-band radiometers which collect background noise data by viewing earth and space. These tests are not complex and can be performed on unmanned vehicles. In the second area, data is obtained on the combined effects of background noise, propagation characteristics (including multipath effects), attitude control dynamics, and typical relative motion kinematics by orbiting promising rendezvous sensor systems which operate with a target vehicle (probably launched from the parent vehicle). These tests are best performed with a man aiding the launch and acquisition of the target. In the third test area, a closed loop rendezvous sequence is performed and sensor operation is evaluated in the presence of Δ V maneuvers as well as the effects tested in the previous area. These tests are beyond the scope of this study since, for the results to be useful, the experiments need to be mission oriented with man in the loop for target acquisition, vehicle attitude control, Δ V corrections, etc. Potential problem areas for the Gemini microwave system include multipath effects and modulation produced by antenna pattern irregularities in addition to the background noise effects.

Both radiation measurement and rendezvous experiments as well as operational rendezvous experience on the Gemini and Apollo programs will provide data of importance. At least some of the experimental results and flight experience from these programs should be assessed before additional experiments are defined.

The suggested tests in the following paragraphs are intended to reflect possible test methods rather than specific equipment.

4.8.3 Suggested tests - Two test methods are suggested, one involving radiometers and the other involving advanced rendezvous sensor systems.

Radiometers - Several narrow-band radiometers are flown on an unmanned, earth-oriented vehicle and mounted to view the earth or space depending on the expected background noise sources for specific rendezvous systems. For this approach, radiometer design parameters such as spectral-band, field-of-view, sensitivity, etc. should be based on rendezvous systems which analysis has shown to be promising. An alternate approach would collect both earth and space background noise data in many bands over a broad spectrum but with relatively small radiometer fields-of-view. The data obtained would then be applicable to a wide range of trackers and sensors. The alternate approach is more complex and might best be performed on a manned vehicle where radiometer pointing and programming can be accomplished manually.

Advanced Rendezvous Sensors - The sensors are assumed to be a laser ranging system and an IR system for angular measurements. The systems are installed on a manned, earth-oriented vehicle which is capable of launching a target vehicle. Either the Gemini or Apollo rendezvous radar serves as a master reference for evaluating the test system's performance. The target vehicle is launched relative to local level-orbit plane coordinates with a ΔV that has principal components in the lateral and radial directions with only a small component along range. The resulting relative motion exercises the test systems acquisition and tracking performance in the presence of both space and earth background noise with increasing displacement along range. Tracking accuracy and the ability to maintain lock-on are evaluated under the above conditions along with the effects of attitude control dynamics and reaction jet characteristics. The rendezvous radar serves as a master measurement reference while a man programs and monitors the experiment.

4.9 Fluid Systems

4.9.1 Objective - The primary objective of this test will be to examine the effects of the space environment on certain areas of fluid technology. Foremost will be the unique problems associated with the pumping of fluids in a zero-g field. The results of these tests will be used to determine the applicable uses of fluids in guidance and control of aerospace vehicles.

4.9.2 Background - Fluid technology is a field which is concerned with storing, transferring, and controlling the characteristics of gases and liquids. The application of this technology to aerospace equipment is not limited to guidance and control equipment. In fact, the guidance and control field is involved in only a small percentage of the total application, which is outlined in Table 4.9-1.

Floated rate and rate-integrating gyros require a fluid which provides the gimbal flotation and acts as the damping system. If the gyro characteristics are to remain constant, the viscous damping characteristic of the fluid must be maintained without change. Temperature compensation is accomplished by using a bellows to adjust internal gyro volume in accordance with temperature change. Operation of these units in space has been proven by flight testing on such programs as Discoverer.

A relatively new advance in the field of fluid technology is the area of fluid amplifiers and computers. These devices are being primarily considered for use on vehicles using hydraulic and pneumatic operated aerodynamic actuators. This results in a control system with less need for signal conversion, e.g. from electrical to pneumatic, etc. Although fluid computers may have application to space vehicles, they are still in the early phases of design. Potential problem areas exist in storage and pumping.

TABLE 4.9-1
FLUID TECHNOLOGY APPLICATIONS

TECHNICAL AREA	VISCOSITY CONTROL	FLUID TRANSFER			PUMP RELIABILITY	HEAT TRANSFER	STORAGE	FLUID MONITORING
		MECHANICAL PUMP	ELECTROMAGNETIC PUMP	PRESSURANT				
Guidance and Control								
Floated Gyro	X							
Fluid Amplifiers/Computers		X	X	X	X		X	
Passive Fluid Flywheels	X							
Active Fluid Flywheels		X	X		X			
Reaction Control Systems				X			X	X
Propulsion								
Fuel Systems				X			X	X
Power								
Fuel Cells		X				X	X	X
Brayton Cycle Systems		X	X	X	X	X		
Environmental Control								
Cooling Equipment						X		X

Passive fluid flywheel control is concerned with damping the precession of a spinning vehicle. It consists of a partially filled, closed fluid loop which is oriented in a plane perpendicular to the vehicle's principle inertia axis. As the vehicle spins, the fluid seeks a location within the closed ring which is a function of the angle between the spin axis and the principle inertia axis. Damping of the relative precession between these axes results.

Active fluid flywheel control employs a closed fluid loop which passes through a pump. The momentum of the system is based on the unit mass of the fluid, the mean radius of the loop from the c.g., and fluid speed. Controlling the speed and direction of the fluid controls the momentum of the system. Several distinct configurations have evolved based on such parameters as type of pump and fluid employed. One of the major advantages of an active fluid flyweeel as a control device is that only at the pumping point is there any specific equipment volume requirement. The loop itself can be made in any arbitrary shape (in a plane) to fit the vehicle.

The fluid must be chosen considering the following factors. It must be compatible with the pump and it must be capable of operating under the temperature environment of space. Although heaters could be employed to overcome temperature problems, an extremely inefficient operation would result. In addition, the fluid should be as heavy as possible to limit maximum required pump speed. Present designs are based on either mechanical or electro-magnetic pumps. While the mechanical method is more efficient, it suffers from low reliability under the space environment due to the number of moving parts and possible leakage problems. Although less efficient, electro-mechanical pumping is very simple; however, a conductive fluid is required. Because of its weight, temperature characteristics, and conductivity, Mercury is considered to be the best fluid for electro-magnetic pumps.

Electro-magnetic pumps function by setting up a fixed magnetic field at some point in and perpendicular to the coil. An electric current is passed through the fluid in a direction perpendicular to both the coil and the magnetic field. The interaction between the electrical and magnetic fields is a force which moves the fluid around the loop. Fluid speed and direction is controlled by the current. A common practice is to keep the loop diameter as small as possible at the pumping point so that the magnetic field can be designed for a minimum size. The primary problems which must be considered when dealing with fluid flywheels are pump efficiency and reliability.

The storage, transfer, and monitoring of liquid propellants are problems which must be considered when using both reaction control and propulsion thrusters. These devices have seen extensive service. The preferred transfer mode is to force the stored liquid from its tankage by introducing a high pressure gas into the tankage behind a membrane which separates the gas from the liquid.

Conventional fuel level and flow monitoring techniques will not function in a zero-g environment. These devices will require orbital testing to prove their capabilities in the operating environment.

Typical applications of fluid technology to the power and environmental control areas are also shown in Table 4.9-1. In addition to the storage and transfer problems, consideration must be given to investigating the technique of transferring heat between a tube and the fluid flowing through it.

4.9.3 Suggested Test - In support of guidance and control requirements, possible tests which should be given consideration include a test of both mechanical and electro-magnetic pumps. The former would be tested for reliability, the latter for efficiency. Following these tests there should be a proof-test of a fluid flywheel system using one of the pumps.

4.10 Velocity/Altitude (V/H) Sensing

4.10.1 Objective - The objective of the experiments will be to evaluate the performance of automatic image velocity sensing systems onboard an orbiting vehicle.

4.10.2 Background - Automatic image velocity sensing using optical techniques has been used for years in aerial photography to compensate for the effect of aircraft motion. The devices use various types of detectors to produce an electrical signal proportional to the image angular rate of motion. The use of these devices for both space vehicle guidance and control is an extension and a refinement of these optical techniques.

Image velocity sensing may be used for space navigation, guidance and control applications. The suggested Landmark Tracking Experiment in Paragraph 4.3 is for an orbital navigation application. Sensors for terminal lunar landing guidance are discussed in References 1 and 2. The V/H sensor used for a reference model in the Landmark Tracking Experiment was designed for image motion compensation in optical camera systems; however, it may also be used for an orbital vehicle yaw attitude control reference. To date, the only known use of image velocity sensing has been for image motion compensation in aircraft reconnaissance systems.

Image velocity is composed of both V/H and space vehicle attitude motion. The velocity, V, is the relative translational velocity perpendicular to the target line-of-sight. The altitude, H, is the slant range to the target. For navigation applications, only V/H is of interest and the sensing system might best be mounted in a stabilized gimbal system to isolate the effects of vehicle attitude motion. For optical camera control applications, since both V/H and vehicle attitude motion are of concern, the sensing system might be mounted directly to the vehicle and since the total image velocity. For guidance applications, V/H is of primary concern; however, vehicle attitude motion is also of interest. Depending on the overall guidance and control mechanization, the sensor may be gimballed or hard-mounted for guidance. For each application, sensor design and vehicle control considerations may permit a hard-mounted configuration or they may dictate a gimballed configuration. Most devices are designed to sense image motion in one axis only; e.g., see Reference 1. The sensor used as a reference in Paragraph 4.3 is mounted in gyro stabilized yaw gimbal and the yaw gimbal is controlled so that the image motion due to V/H occurs around one axis.

4.10.3 Suggested Tests - Both visual and infrared imaging techniques have been suggested for use in an image velocity system. While high altitude aircraft flight testing can provide performance data on the techniques, orbital testing is necessary to assess the individual and combined effects of cloud cover, target contrast, aspect angle, and orbital V/H. Orbital testing can ultimately be conducted on a variety of devices. However, the test suggested in paragraph 4.3 would provide a useful initial test. A test of the V/H sensor as a yaw attitude reference would also be worthwhile.

REFERENCES

1. Hultquist, P.F. and Bartoe, O.E. Jr., "A Slot and Bar Scanner for Measuring Velocity," IEEE Trans. ANE, December 1964.
2. M.A.C. Report 7931, Volume III, Part 2, "Surveyor Spacecraft," 15 December 1960.

4.11 Control Logic

4.11.1 Objective - The objectives of this experiment are to evaluate control logic techniques in the orbital environment and obtain data for correlation with ground test results.

4.11.2 Background - Control logic is used on orbiting vehicles to simultaneously minimize attitude rates and fuel consumption of reaction control systems. The basic object of the technique is to provide a minimum pulse which will return the vehicle to within the deadband limits at a minimum attitude rate. Methods for accomplishing this objective vary from pre-set logic to forms of adaptive logic. Pulsing techniques include pulse duration modulation and pulse repitition modulation. One suggested pulse modulation technique measures the time between opposing jet pulses (time in deadband) and reduces the succeeding pulse duration until a minimum pulse width is achieved. For pulse repetition frequency modulation, a minimum pulse width is repeated until the vehicle is within the allowable deadband limits. Although the goal is minimum rates, most control methods operate on attitude data because of the inaccuracies in rate sensors at the desired low rates.

Predicted fuel consumption for a given vehicle mission is based upon simulation and computer tests. The comparative penalties of added weight and fuel depletion prior to accomplishing the planned mission are weighted in order to arrive at a proper safety factor which is used to determine the vehicle fuel capacity requirements.

4.11.3 Recommended Tests - Since the errors in predicting fuel consumption can be quite large and the parameters for determining the proper safety factor are generally incompatable, orbital tests are suggested as a means of improving the accuracy of predicting fuel consumption. These tests would provide a means of comparing predicted fuel consumption with measured fuel consumption. Active control of the carrier vehicle would be required for the test. Several control logic techniques could be evaluated in one test. In addition, these tests could be instrumented to measure reaction jet performance and the accuracy of sensors designed to measure fuel supply in a zero-g environment.

4.12 Reaction Jets

4.12.1 Objective - This experiment will demonstrate the operation and performance of reaction jets in the orbital environment. In addition, data will be obtained for correlation with ground test results.

4.12.2 Background - Reaction jets constitute the most commonly used space vehicle momentum exchange system. These are divided into two categories: cold gas jets, which expel high pressure gas that is stored on board the vehicle; and hot gas jets, which expel gas that has been created by burning liquid propellants or combining a liquid with a catalyst.

Cold gas jets are quite simple. The system consists of a gaseous propellant supply stored under pressure, a regulator, a valve, and a nozzle. Typical gases employed are hydrogen and nitrogen. The regulator reduces the gas pressure from the storage valve to the thrust valve and the valve releases the gas when thrust is required. The maximum thrust level remains constant within the accuracy of the regulator. The minimum impulse repeatability depends on both the regulator accuracy and the mechanical time constant of the valve. The response time for cold gas jets is the shortest for any reaction jet, limited only by the valve time constant. Disadvantages of cold gas systems are low specific impulse, which is a weight disadvantage, and low density, which is a volume disadvantage. These factors prohibit the employment of cold gas jets in many space systems.

Hot gas jets can be classed as mono-propellant and bi-propellant. Either of these systems consists of propellant tankage, valves, a combustion chamber, and a nozzle. The mono-propellant system functions by decomposing the fuel through contact with a catalyst or application of heat in the combustion chamber. Typical propellants are 90 percent hydrogen peroxide and anhydrous hydrazine (N_2H_2). A common catalyst used with hydrogen peroxide is silver and the chief product of the decomposition is steam. The delay associated with propellant decomposition is the primary liability of this system. The total thrust level is dependent on fuel flow and efficiency in the decomposition process. The ignition and tail-off characteristics are determined by valve operation and the initial and final reactions of the fuel with the catalyst. None of these characteristics are comparable in accuracy and repeatability to the cold gas system, nor is the response time as rapid.

The bi-propellant hot gas, or hypergolic, system employs a liquid fuel and an oxidizer which ignite when they are mixed in the combustion chamber. Typical fuels are hydrazine and UDMH ((CH_3 NNH_2). Both use nitrogen tetroxide (N_2O_4) as an oxidizer. The response time of the hypergolic system depends chiefly on the valve time constant which makes this system almost as fast as the cold gas system. Ignition and tail-off characteristics and thrust level repeatability depend on both valve operation and combustion efficiency. In neither case is the hypergolic system as satisfactory as the cold gas system. Hot gas systems require tankage pressurizing to force the fuel flow in zero-g conditions. Even with this considered into total system size and weight, hot gas systems are considerably smaller than cold gas systems for an equivalent total thrust. Effeciency of propellant consumption is also much higher for hot gas systems.

4.12.3 Suggested Tests - The majority of testing required to develop and prove a reaction jet can be performed on the ground. However, determination of the jet thrusting characteristics in a vacuum environment is hampered by the difficulty of maintaining the pressure in the presence of a mass expulsion device. Orbital testing will provide a final proof test of the system and provide the data necessary to prove the validity of ground derived test results.

4.13 Extravehicular Control

4.13.1 Objective - This experiment will evaluate the operation of a tethered extravehicular payload system in which tether control is from either the payload or the parent space vehicle. Results of this test would be useful in planning further ground tests and full-scale flight tests for astronaut extravehicular activities on the Gemini and Apollo programs.

4.13.2 Background - As space flight evolves, increased emphasis will be placed on operations external to a space vehicle. These external operations may involve either an astronaut or sensing equipment as the primary extravehicular payload.

Most current concepts of astronauts in free space have them equipped with a propulsion and life support system capable of sustaining him in the high vacuum of free space for limited time periods. His propulsive capability allows the astronaut to depart to great distances from the parent vehicle. In this regard, careful consideration must be given to procedures for aiding the astronaut in the event of malfunction of the propulsion and life support system during some extravehicular maneuver. Further, while physically separated from his carrier vehicle, the astronaut may be incapacitated, or may be subject to hazards that would hinder or prevent his return, thus requiring an emergency retrieval system. A physical connection between the astronaut and his carrier vehicle is necessary to provide a back-up or alternate retrieval system, as well as to provide a sense of well being to the astronaut during extra-vehicular operations. These requirements can be satisfied by employing a controlled tethering/retrieval system. However, studies of such systems have indicated the existence of some dynamic problems which could result in injury to the astronaut. For this reason, it becomes imperative that the devices be thoroughly tested prior to use by astronauts.

Various extravehicular sensing equipment, such as the "dog-on-a-leash" guidance concept, have been proposed. In many cases it is desirable for the sensing equipment to remain in close proximity to the parent vehicle either for data return or for return of the payload package. The payload package can be tethered to satisfy these requirements in which case a tethering control and/or retrieval system is indicated.

Analytical studies to date on tethering control have had to assume a completely rigid system and model tests are limited to two-dimensional (three-degree-of-freedom) evaluations. There are four major areas which should be investigated more thoroughly in space:

a. Six degree-of-freedom body dynamics.

b. Tether vibration modes (linear and angular).

c. Sensing element accuracies.

d. Prime mover and reel evaluation under space conditions.

The first two areas are considered the most important since they cannot be conclusively evaluated in ground tests and the information is essential to early implementation of long-range manned extravehicular activities. Six degree-of-freedom simulation of controlled tethering is difficult in a one-g field, especially with manned tests. No simulation facilities have been built or are planned that would acceptably simulate controlled tethering.

4.13.3 Suggested Test - The Marquardt Corporation has proposed to evaluate a model of a three body astronaut retrieval unit (Reference 1). This system automatically senses the tether tension plus the distance, rate and angle of the tethered package relative to the parent vehicle. With this information, the system regulates the package retrieval using a small electric motor and reel mechanism. A third body anchor mass is also deployed to minimize the increase in package angular velocity as it is reeled in.

The test could best be performed on a manned vehicle so that the astronaut can program the experiment, observe the results, and report on the experiment. If performed on an unmanned vehicle, major problems include the deployment sequence and methods of keeping the tether separate from extendable booms, antennas, etc. during both deployment and retrieval.

REFERENCE:

1. The Marquardt Corporation PD Study No. 666, "Three Body Tethering Evaluation," (3 pages), October 1964.

4.14 Passive Control Techniques

4.14.1 Objective - The objective of this experiment is to demonstrate performance and obtain design data for passive satellite attitude control systems based on the use of solar radiation pressure, the earth's magnetic field, or aerodynamic forces for vehicle orientation.

4.14.2 Discussion - There are four major natural sources of torque - solar radiation pressure, gravity gradient, the earth's magnetic field, and aerodynamic forces. All four of these natural forces can be utilized for control. The gravity gradient effect and its use as a source of torque are discussed in Paragraphs 2.3 and 3.1 and will not be repeated here. The remaining three sources are discussed in the following paragraphs:

Solar Radiation Pressure - Bombardment of the satellite surfaces by photons from the sun creates forces on the spacecraft that are determined by the surfaces' reflective properties. When the center of radiation pressure through which these forces act coincides with the center of mass, the vehicle is caused to rotate.

The radiation power in the vicinity of the earth is 1.94 calories per sq. cm per min., corresponding to a pressure of 9.4×10^{-8} lb. per sq. ft. for complete absorption, and can provide useful control torques if the spacecraft is equipped with a large movable reflective sail. Such a scheme is hard to implement and requires special control during eclipse, and momentum storage during periods when torques about required axes cannot be obtained. An alternate method would be to obtain restoring torques from focused radiation. A reflector and collector could be used in a purely passive manner to provide a useable force. Vehicle control using solar radiation pressure is only of value in extremely high altitude or interplatetary missions.

Earth's Magnetic Field - The earth's magnetic field has already been used with magnetic rod arrays for despinning Transits 1B and 2A. This work and other studies show that a complete control system based on obtaining moments by energizing satellite-fixed current carrying coils is feasible. For lower altitudes the torque can be obtained with less power on lighter coils.

For satellites more than 100 miles up, circulating currents in the atmosphere and surface irregularities do not significantly affect the approximation of a field produced by a simple dipole at the earth's center. The axis of the dipole is skewed at an angle of about 18 degrees with respect to the earth's spin axis, so the dipole axis and the field precess around the spin axes. Only in a 24-hour orbit that contains the dipole is it impossible to generate torques for complete three-axis control. In all other cases it appears possible to generate control torques about all three axes, but the torques may not be available at the required instant, which suggests need for momentum storage. Knowledge of field magnitude and direction, or both, is essential to proper energizing of the coils and must be supplied by either in-flight computation or measurement.

Use of the earth's magnetic field for stabilization is limited since the torque generated is always about an axis perpendicular to the earth's field, and therefore only two axes can be controlled at once. Devices such as inertia fly-wheels or gyros must be added to store the momentum along the axis or axes that cannot be controlled until the spacecraft reaches an orbit position where the momentum can be magnetically transferred from the uncontrollable axis. Since it is difficult to achieve adequate control torques at altitudes above 10,000 miles, the method is limited to lower altitudes.

Aerodynamic Torques - Vehicle stabilization by aerodynamic forces can be attained at altitudes intermediate between the minimum for reasonable life-time

(80 n.mi.) and the maximum for such forces to be usable (300 n.mi.) For example, the magnitude of the deceleration reported on Sputnik III at perigee (226 km) is cited in Reference (1) as 3×10^{-6}g. Aerodynamic forces are considered significant up to 300 miles altitude, where the dynamic pressure drops to 4.4×10^{-8} lb./ft.^{2}.

Natural vehicle stability is obtained by insuring that the center of pressure lies behind the center of mass relative to the direction of travel. However, the aerodynamic forces provide no damping. Damping may be obtained by coupling the satellite to an external drag device by a flexible joint. The drag device could be a balloon (sea anchor concept) or box kite acting as a rudder.

Aerodynamic forces and moments are strongly dependent on vehicle shape, in particular the distance between the aerodynamic surface and the center of pressure. Careful design is required to provide maximum aerodynamic stability and minimize the effects of destabilizing gravity gradient torques as pointed out in Reference (2).

4.14.3 Recommended Tests - Further investigation of methods of implementing passive control systems associated with the natural phenomena of solar radiation, earth magnetic field and atmosphere is required before any recommendations for specific tests can be made.

REFERENCES

1. Sedov, L. I., "Dynamic Effects of the Motion of Earth Sputniks", Proc. IXTH IAF Congress, Amsterdam, August 1959

2. DeBra, D. B., and Stearns, E. V., "Attitude Control", Electrical Engineering, Vol. 77, 1958, Page 1088

4.15 Space Environment Tests

4.15.1 Objective - Orbital testing would provide data to determine the characteristics of several guidance and control devices operating with the combined space environment. Particular interest centers on zero-g and continuous or intermittent operational duty cycles for long orbital durations. The data can be used to evaluate the effectiveness of ground tests.

4.15.2 Background - The space environment is characterized by the combination of various levels of pressure, gravitational forces, and thermal and particle radiation. The environment is discussed in detail in Paragraph 3, Volume I. Ground simulation techniques have not advanced sufficiently to allow simulation of the total space environment.

Simultaneous simulation of environments such as zero-g, low pressure; and particle radiation can be performed for only short time periods in ground tests. Ground test evaluation of lubricants and lubricant feed techniques is seriously hampered by the inability to satisfactorily simulate the zero-g environment for long time periods; therefore, long term testing must be confined to the actual orbital situation.

4.15.3 Suggested Tests - Many devices used in guidance and control systems such as floated gyros, fluid flow systems, extendable booms, solar panels, gimballed systems, rotating devices with bearings, etc. are expected to operate in a different manner in the zero-g space environment than in the ground laboratory conditions. If the devices could be evaluated in the zero-g environment and optimized for that condition, design changes could be made which would reduce power and cost. Calibration problems will still require continued efforts following optimized design.

A typical long term test would determine the drift rate of several types of gyros with and without temperature controls and with various operating conditions, such as; continuous or intermittent operation; reactivation after shut-down for a long period of time; high and low spin motor voltage. Instrumentation would be primarily to measure drift rate as a function of time. Units could be exposed to radiation, micrometeorites and solar thermal radiations to determine the effect.

Other devices which would benefit from long term orbital testing include: electronic components, with emphasis on semi-conductors; optical surfaces; inflatable structures, such as antennas; temperature control devices; and special bearings such as gas, electrostatic and jewel.

APPENDIX A

MASTER ATTITUDE REFERENCE SYSTEM CONSIDERATIONS

A.1 Introduction - The purpose of this Appendix is to describe techniques and equipment capabilities of master attitude systems to facilitate selection of an optimum system for each category A and B experiment. Most of the experiments require determination of vehicle attitude when oriented either inertially or to earth local vertical. The sensors discussed in this appendix may be used individually or in combination to provide the desired reference, depending upon the specific requirements of each experiment.

A unique situation exists for those experiments which require an earth oriented vehicle and a precision attitude determination, such as the Horizon Definition, Horizon Sensor Accuracy, Gravity Gradient Sensor and Gyro compassing experiments. The precision requirement can only be met by use of a stellar reference. An OAO type of tracker on an earth local vertical oriented vehicle provides the precision attitude determination required. A star tracker may also be used for those experiments on inertially oriented vehicles such as the Star Characteristics and Electro-Static Gyro experiments.

A lower precision earth reference, obtained by using a horizon sensor and gyrocompassing, may be used for the Ion Attitude Sensing, High Reliability Horizon Sensor and Gravity Gradient Controls - Passive Damping experiments.

A.2 Attitude Determination, Two Axes - Several techniques are available for determination of a vehicle's attitude relative to a set of coordinate axes. A general discussion of several of the various methods is contained in the following paragraphs.

Horizon Sensors (for measurement of direction of local vertical relative to vehicle) - A horizon sensor system provides a measure of vehicle planar attitude with respect to the earth local vertical axis. Most horizon sensor systems have outputs referred to as pitch and roll, however, the outputs are subject to cross coupling and are not normally independent of each other or of vehicle attitude with respect to the orbit plane. As discussed in detail in Experiment 2.5, Horizon Sensor Accuracy, horizon sensors are susceptable to errors caused by earth atmospheric anomalies. The errors may be from 0.1° to greater than 1°, depending upon the specific sensor design. For experiments requiring an earth local vertical reference with an accuracy of 3° the horizon sensor represents the smallest, lowest powered unit available with proven long mission capabilities.

Sun Sensor/Sun Tracker (for measurement of direction of sunline relative to vehicle) - Two coordinates of a vehicle's attitude, referenced to inertial space, may be determined by use of a sun sensor tracker. A no-moving-part sensor is capable of accuracies of 0.5° or better about null. A sun tracker is capable of greater accuracy, but as accuracy increases, so does complexity, power and weight. Both sun sensors and trackers are susceptible to errors of up to 0.1° introduced by sun-spots causing a variance in the light centroid of the sun. A sun tracker or sensor functions only during that portion of the orbit when the vehicle is on the sun-lit side of the earth, or slightly over 50% of the time (depending upon the orbit parameters).

Star Tracker (for measurement of direction of star relative to vehicle) - A star tracker will provide a high precision measurement in two axes of vehicle attitude with respect to an inertial set of coordinate axes. A star tracker will provide almost full orbit coverage if several different stars are tracked sequentially. For some altitudes and orbit inclinations, it is possible that a single star can be selected that will remain visible for the entire orbit.

Electrostatic Gyro (ESG) (for measurement of deviation from reference vehicle attitude) - An electrostatic gyro provides an inertial attitude memory reference in two axes. For long term applications the ESG has the disadvantage of lowered precision due to drift (expected to be extremely low in an orbital environment). The ESG has the advantage over electro-optical sensors that an optical field-of-view or window is not required; hence, mounting and vehicle design is less critical. ESG units have received extensive ground tests; however, orbital evaluation of the gyro drift characteristics is needed (see Paragraph 2.1, Electro-Static Gyro).

A.3 Attitude Determination/Memory - Three Axes -

Strapped-Down Platform - Three rate integrating gyros mounted directly to the payload can provide a three axis reference. This system is limited to short term applications by random drift rates in the order of 0.1 to 1 degree per hour. Also, the gyro reference must be initially aligned to some known reference. Operational problems include the relatively high power consumption and the limited input freedom ($\pm 10^{\circ}$). If the payload is controlled to an inertial orientation, the constraints imposed by limited input freedom are usually not serious. However, if the payload is to be controlled to an earth reference, the pitch gyro is torqued from an external reference to prevent hitting the stops.

Strapped-Down Platform/Horizon Sensor - In order to overcome drift for long-term operation, an external reference system may be utilized in conjunction with the strapped-down rate-integrating gyros. For pitch and roll, the reference is a horizon sensor system. For yaw, the reference is a gyrocompass. The accuracy of this system is limited primarily by the accuracy of the horizon sensor.

Gimballed Platform - The gimballed platform configuration can use three rate integrating gyros or two two-degree-of-freedom gyros mounted on the inner element of a gimballed system to provide a three axis reference. The gyros can be higher quality (0.05 to 0.5 degrees per hour random drift) than those associated with a strapped-down platform since the gimbals are used to isolate payload motion from the gyro and wide gyro input freedom is not necessary. However, a gimballed system is more complex and power consumption is higher than the strapped-down system. In many ways the gimballed approach is similar to the strapped configuration; initial alignment is required, and the gyros can be torqued from an external reference such as a horizon sensor or a star tracker. For the attitude reference function, accelerometers need not be mounted on the inner element, thus making the platform less complex than a full inertial platform.

Gimballed Platform/Horizon Sensor - As with the strapped-down platform configuration, if the pitch and roll gyros are torqued or aligned with a horizon sensor and gyrocompassing is used for yaw alignment, an earth orbit plane reference system is established. The accuracy of this system is primarily limited by the inaccuracies associated with horizon sensing and gyrocompassing. The ultimate accuracy of the system cannot be determined until the horizon sensor in-orbit accuracy is determined; however, it is generally agreed that accuracies of better than 1° in pitch and roll, and 5° in yaw, are obtained with the platform-horizon sensor configuration.

Star Tracker/Two-Axis Gyro (for determination of direction of local vertical relative to vehicle) - A star tracker in conjunction with a conventional two-axis free gyro may be used to provide a two-axis earth local vertical reference system of high precision. The vehicle precision attitude is computed through use of the coordinates to two tracked stars, the time base, vehicle orbit position (as determined by ground tracking), gyro memory and the star tracker gimbal angles. A horizon sensor is used for initial determination of vehicle attitude in order that the specific tracked stars may be established by data reduction techniques. The operational and data reduction technique is complex; however, a man is not required to be actively in the loop.

The star tracker is programmed to a position near the forward gimbal stop and a search pattern is initiated. When a star is detected, the system is commanded to acquire or lock-on to the star. The two axis gyro is slaved to the star angle line-of-sight and switched to an inertial mode. The star tracker lock-on is broken and the search/acquire/lock-on sequence is repeated for a second star. The second star should be separated from the first star by an angle of at least 30°. The system continues tracking the second star until the telescope line-of-sight reaches a predetermined position to the aft of the vehicle. The gyro is slaved to the second star line-of-sight and switched again to inertial mode. The star tracker is slewed to a forward position and acquires a third star. The cycle of gyro slaving, searching and acquiring is repeated until a command is received for a standby or an off condition.

The total search and track area is within a field-of-view cone of 150° for an OAO type star tracker. The field-of-view, with the central axis of the cone aligned to the vehicle yaw axis, is illustrated and discussed in Paragraph 2.4. If a star is acquired at the forward position and tracks through the majority of the 150° angle, and if the vehicle alignment is maintained at an earth local vertical orientation, the tracking time will be approximately 22 minutes.

Although the star tracker and gyro outputs are not a direct measure of the vehicle attitude with respect to earth, the earth reference information can be obtained. A precision measure of the spacecraft attitude with respect to local vertical requires using time as a baseline, the star tracker angles, the vehicle position in orbit and the star catalog position of the tracked star. For any given position in space at any instant in time, the angular coordinates to the stars and planets is very precisely known. For a near-earth orbiting vehicle which is local earth vertical oriented, any deviation between the star tracker and gyro angles and the known coordinates to particular stars is a direct measure of the spacecraft attitude with respect to local vertical.

Figure A-1 is a data reduction flow diagram. As shown in the figure, using the horizon sensor outputs, the vehicle's position in orbit (established by ground tracking), star tracker gimbal angles and a Boss General Catalog of the Stars (or equivalent) the specific star being tracked is determined by computation.

DATA REDUCTION FLOW DIAGRAM
TWO AXIS ATTITUDE DETERMINATION, EARTH REFERENCE

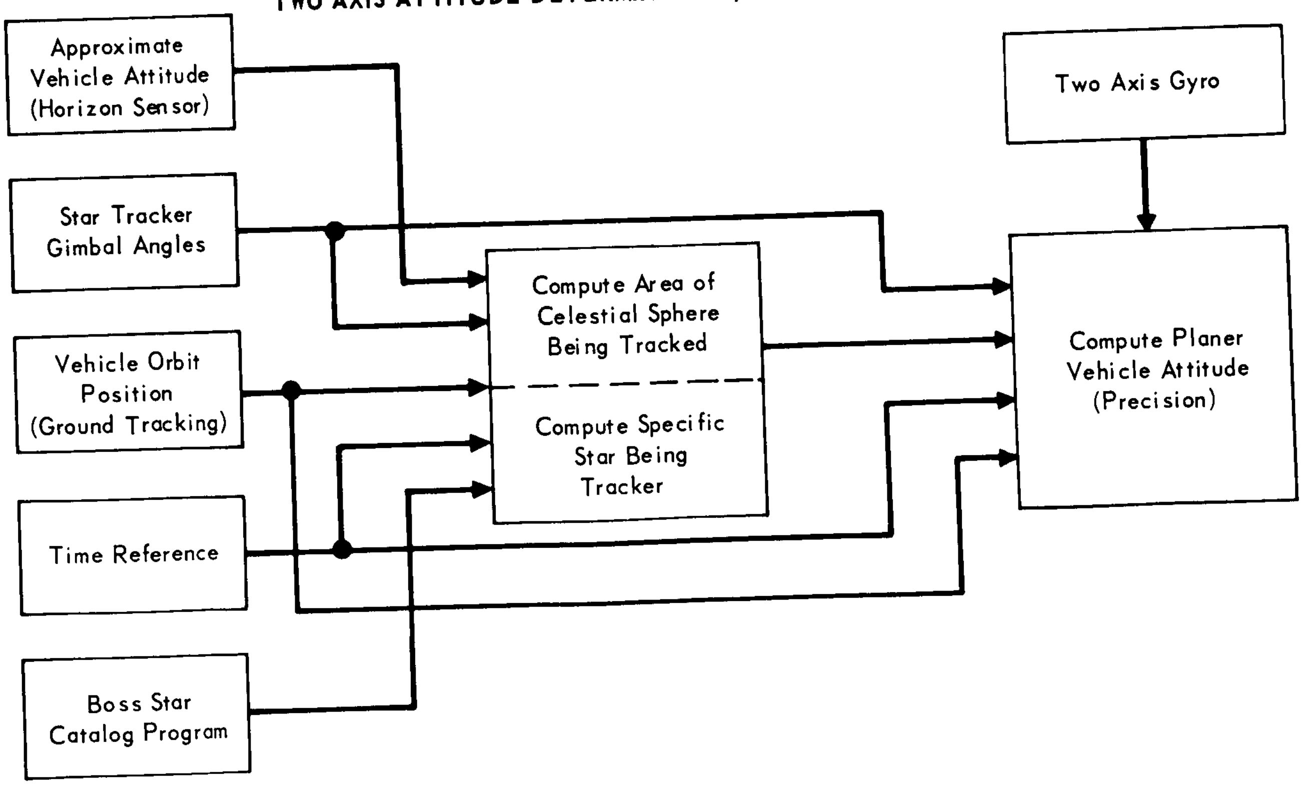


FIGURE A-1

The two-axis gyro serves as an inertial memory for the co-ordinates of the first star while the star tracker provides the co-ordinates to the second star. The vehicle precision attitude is computed through use of the co-ordinates to the two stars, the time base, and the vehicle orbit position.

An Electrostatic Gyro (ESG), with its projected low drift rates, would provide a nearly ideal inertial memory unit for the star tracker/two-axis gyro reference technique. The gyro spin axis could be aligned to a star and re-alignment (or measurement of deviation angle) would not require repeating for several hours if an accuracy of 0.05° is desired. Standard two-axis gyros with drift rates of 0.2°/hour will require realignment about once each 15 minutes to provide a similar performance capability.

The major sources of inaccuracy in this technique are the star tracker readouts, gyro slaving to the star tracker line-of-sight and the gyro drift. Typical error numbers are 20 to 30 arc seconds for the star tracker, 60 to 80 arc seconds for gyro slaving (including mounting), and 0.2°/hour for gyro drift. Assuming the above errors are independent, it is reasonable to expect a definition of the horizontal plane to an accuracy on the order of 0.05°. An ESG would reduce the drift to less than 0.01°/hour which would provide an improvement in the accuracy of horizontal plane determination.

Gimballed Platform/Star Tracker - In order to obtain a relatively long term inertial reference in three axes, a combination of a gimballed platform and a star tracker may be used. The platform is inertially aligned while the star tracker measures the angle to one of the navigation stars. In order to reference the data to a known set of co-ordinates, either a sun reference, an earth reference, or a measurement to a second star is needed. A method using the star tracker would be to slow the star tracker to search for and acquire a second navigation star. The platform serves as an inertial memory of the co-ordinates to the first star. After the second star is acquired, the platform gimbal angles and the star tracker gimbal angles are read out. This provides a simultaneous measurement of the co-ordinates to two stars if it is assumed the drift rate of the gyros is low (gyro drift is a major error source when using this technique). The major problem associated with this technique is the search for and acquisition of specific stars. Sufficient stored stellar co-ordinates must be included in the computer to cover all of the planned orbit.

Use of the two-star technique provides enough data to derive vehicle attitude in three axes with respect to earth local vertical if ground tracking is utilized. Platforms are life limited (wear out problems); therefore, this configuration is not usable as a long term master reference.

<u>Electro-Static Gyro/Star Tracker</u> - Two electro-static gyros (ESG) and a star tracker may prove to be a partial answer to the difficulties associated with the platform-star tracker configuration. Orbital evaluation of the ESG is needed to determine its drift and life characteristics. If the drift of the gyro is as low as predicted, and if run-down time is as long as expected, the ESG-star tracker combination could well prove to be a nearly ideal, long-life inertial reference. The ESG and star tracker are both capable of high precision performance; hence, accuracies within 0.01° should be within the state-of-the-art.

<u>Multiple Star Trackers</u> - Two or more identical star trackers are used, each searching and tracking different sectors of the star field. Two star trackers, simultaneously locked onto stars widely separated, provide the necessary information from which individual pitch, roll and yaw axis co-ordinates with respect to local vertical could be obtained to an accuracy of about $.02^{\circ}$ or better.

The star trackers would be pointed to predetermined guidestar positions in the celestial sphere, with at least 40 to 60° separation between the two telescope angles. The two star trackers would search until acquisition and track is obtained. The star tracker gimbal angles provide the measure of vehicle attitude. This technique has the advantage of high accuracy and three-axis information being readily obtained from ground computers. The two star tracker technique has the disadvantages that:

a. Weight volume and power is greater than some other techniques.

b. Installation and alignment is difficult.

c. The probability of tracking two stars essentially full time is lower than for one star tracker tracking one star.

d. Since two stars are being tracked, the difficulty of ascertaining which particular stars are being tracked is doubled over a single star technique.

Star Field Correlators/Mappers - A modified version of the Project Scanner star field mapper, a wide field of view mapper such as proposed by Nortronics, the ATL Stellar Attitude Measuring System (SAMS) or any one of several star field mappers could be used for measuring vehicle attitude. These units will, in general, provide data on the coordinates to more than one star, hence three axis information may be derived.

The Project Scanner star field mapper, built by Baird Atomic, requires a spinning vehicle for operation. The unit could be used on a stabilized vehicle if the spinning motion were imparted to only the sensor rather than the entire vehicle.

The Nortronics star mapper is apparently within the state-of-the-art insofar as hardware is concerned. However, further development and flight test is required prior to using the system as a master attitude reference.

TABLE A-1
GENERAL CHARACTERISTICS OF ATTITUDE REFERENCE SYSTEMS

TYPE OF SYSTEM	INERTIAL OR EARTH	TYPICAL SIZE CU.FT.	PARAMETERS WEIGHT POUNDS	RELATIVE PRECISION	LONG OR SHORT TERM (DAY VS HOURS)	RELIABILITY (1 MONTH OR MORE)	DEVELOPMENT STATUS	REMARKS
Two Axis Horizon Sensor	Earth	0.15	15	0.1° to 1° (to be established)	Long	Good	Good	Ultimate accuracy is unknown – Orbital evaluation needed.
Sun Sensor/ Tracker	Inertial	0.2	20	0.1°	Long	Good	Fair Good	Further development required.
Star Tracker	Inertial	1.4	45	0.008° or	Long	Good	Fair	
ESG	Inertial	0.4	20	0.01° Low drift (to be established)	Medium to Long	Good	Poor	Orbital evaluation of drift rate is needed.
Three Axis Star Tracker & Two Axis Free Gyro	Earth – Inertial	1.55	60	.05° or better	Long	Good	Fair	Data reduction is complex.
Strap-down Integrating Gyro	Inertial	0.5	20	0.1° (Subject to Drift)	Short	Fair	Good	Drift is a major problem and limits usefulness.
Strap-down & Horizon Sensor	Earth	0.65	35	0.3° (Limited by Horizon Sensor)	Medium	Fair	Good	
Gimballed Platform (Gyros only)	Inertial	0.8	35	0.1° (Subject to Drift)	Short	Poor	Fair	Drift is a problem – Life is relatively short.
Platform and Horizon Sensor	Earth	0.95	50	0.3° (Limited by Horizon Sensor)	Medium	Poor	Fair	Life is limited by Platform.
Platform and Star Tracker	Inertial	2.2	80	0.01°	Medium	Poor Limited by Platform	Fair	Extensive usage on aircraft – Data reduction is complex – Needs orbital evaluation – Life limited by platform.
Star Tracker ESGs	Inertial	2.2	85	0.01°	Long	Good	Poor	Needs further development.
Star Tracker – (Multiple)	Inertial	2.8	90	0.01° or better	Long	Good	Fair	To be used on OAO – Data reduction and correlation is complex.
Star Mapper	Inertial	0.8	20	0.1°	Long	Good	Poor	Needs further development and orbital evaluation.

The Advanced Technology Labs SAMS is under development for NASA to be used in measuring vehicle attitude. Present plans are to design the unit for 0.1° performance. Three stars are required in the field-of-view to provide usable signals. An analysis of the three-star patterns available in an equatorial orbit indicates a high probability of operation throughout an orbit. A similar analysis is required for polar orbits.

Other proposed star mappers also suffer from lack of development and flight experience. Most of the proposed units are designed for performance accuracies in the order of 0.1°. Improved performance seems possible; however, further study is needed to determine the ultimate accuracy of each design.

A.4 Equipment Capabilities - Table A-1 is a tabulation of general characteristics of the two and three axis reference systems previously discussed. Both inertial and earth reference systems of varying degrees of accuracy are shown in the table. The values given for size and weight are approximate and specific unit values will vary from those given. Generally accepted performance characteristics are given for relative precision of attitude determination, although improvements in performance can be expected in the future. The long or short term column provides an indication of whether the design concept is applicable to a short term reference of hours duration or to a lengthier mission duration of days, perhaps weeks. The relative reliability indication is based on a one month mission requirement. Proposed laser and microwave reference systems, as well as many other types of systems, have not been included in Table A-1. These other systems are not listed because of development status, unknown or unproved capability or because of weight and power penalties greater than for the equipment listed.

A.5 Selection Considerations - The selection of a master attitude reference system for an experiment requires an evaluation and trade off analysis of several parameters. The major considerations for selection of the system are:

a. Level of precision required.

b. Development status of the systems under consideration.

c. Relative size, weight, power and cost.

d. Complexity of data reduction associated with the reference system.

e. Complexity of data retrieval techniques required with the reference system.

f. Time duration of the planned experiment.

g. Required reliability of the reference system and the overall experiment.

h. Carrier vehicle and it's on-board attitude reference system.

i. Probability of the reference system functioning without a man actively in the loop, i.e., an unmanned carrier vehicle.

These considerations are not necessarily in the order of priority. Trade-off studies and detailed analysis are required to arrive at an optimum system for each experiment.

Of prime importance in the selection of a reference system is the required measurement accuracy, the expected time duration of each experiment, hardware availability and development status. Electrostatic Gyros (ESG) appear promising as a part of a long term reference system; however, further development effort and orbital testing is required to measure their capability. The ESG has the advantage of low drift rates as compared to rate-integrating and free gyros and no field-of-view requirements such as required by electro-optical sensors. Gimballed platforms are available; however, platforms tend to be complex, unreliable and have excessive drift rates for long term applications. Star mappers of several types are being actively developed and will be available in the near future. Star trackers which are space qualified are in production for the Orbiting Astronomical Observatory and will be flight proven in the very near future.

For this study, prime consideration for selection of a master attitude reference was given to the orbit parameters, required vehicle stabilization, operating duration of the experiment and the required accuracy. Table A-2 is a summary of the Category A and B experiment requirements and the recommended type of master reference. Two of the experiments require a star tracker on an inertially oriented vehicle. Four of the experiments require a star tracker operating on an earth oriented vehicle to provide precision local vertical information. Three of the experiments require an earth reference system such as horizon sensing and gyrocompassing, and five of the experiments do not require a master attitude reference. Table A-3 shows the Category A and B experiments grouped by master reference selections. The grouping indicates the possibility of multiple experiments sharing a common master reference; however, further trade-off studies on the effect of variations of the stabilization and orbit parameters on each experiment must be conducted to evaluate this possibility.

TABLE A-2
SUMMARY OF EXPERIMENT MASTER REFERENCE REQUIREMENTS

EXPERIMENT	ORBIT PARAMETERS h (NM)	ORBIT PARAMETERS e	ORBIT PARAMETERS i (DEG.)	STABILIZATION ORIENTATION	STABILIZATION ATT. (DEG.)	STABILIZATION ATT. RATE (DEG./SEC.)	TEST TIME DURATION (NOM.)	TEST TIME OPERATE (NOM.)	MASTER ATT. REFERENCE ACCURACY REQUIRED	MASTER ATT. REFERENCE SELECTED REFERENCE
Category "A"										
1. Electrostatic Gyro	> 200	any	any	Solar-inertial or pure inertial (3 axes)	± 2	≤ 0.05	30 days	28 days	0.008°	Star or Sun Tracker
2. Low g Accelerometer	> 300	< 0.01	any	Earth in p & r orbit plane in y	± 5	y ≤ 0.0005 r & p ≤ 0.05	> 1 mo.	5 days	–	None
3. Gravity Gradient Sensor	300 to 500	< 0.01	any	Earth-orbit plane (3 axes)	± 0.1	≤ 0.05	3 days	1.5 hr.	0.005°	Star Tracker
4. Earth Horizon Definition	100 to 600	< 0.01	> 70	Earth-orbit plane (3 axes)	y < 5 p & r < 1	≤ 0.1	2 wks.	9 hr.	0.008°	Star Tracker and Two Axis Gyro
5. Horizon Sensor Accuracy	200 to 300	< 0.07	> 70	Earth-orbit plane (3 axes)	y < 5 p & r < 1	≤ 0.1	2 wks.	9 hr.	0.008°	Star Tracker and Two Axis Gyro
6. Gas Bearing Performance	> 200	any	any	any	any	≤ 0.1	4 days	3.3 hr.	–	None
7. Star Characteristics	200 to 500	any	any	Pure inertial (3 axes)	± 0.1	≤ 0.1	3–6 mo.	20 hr.	0.008°	Star Tracker
Category "B"										
1. Gravity Gradient Controls-Passive Damping	300 to	< 0.01	any	Earth-orbit plane (3 axes)	± 30	≤ 0.1	1 day	20 hr.	0.5°	Horizon Sensor & Gyrocompass (or Sun Sensor)
2. Ion Attitude Sensing	150 to 300	< 0.01	90	Earth-orbit plane (3 axes)	± 10 in p & y ± 3 in r	≤ 0.1	1 wk.	1 wk.	0.1°	Gyrocompass
3. Gyrocompassing	150 to 300	< 0.01	any	Earth-orbit plane (3 axes)	± 2	≤ 0.1	1 wk.	5 days	0.008°	Sun Sensor or Star Tracker
4. High Reliability Horizon Sensor	200	< 0.01	> 70	Earth (2 axes)	± 1	≤ 0.1	3–6 mo.	9 hr.	0.1°	Horizon Sensor (0.1° Acc.)
5. Star Recognition	> 200	any	90	Pure inertial (3 axes)	± 1°	≤ 0.05	3 days	9 hr.	0.008°	Star Tracker Horizon Sensor
6. Small Impulse Devices	> 150	any	any	any	any	≤ 0.1	2 days	50 min.	–	None
7. Optical Windows and Mirrors	Van Allen Belts	any	any	Solar (2 axes)	± 30	any	6 mo.	9 hr. mo.	–	None
8. Bearings and Lubricants	any 600	any	any	any	any	any	> 6	14 day mo.	–	None

TABLE A-3

GROUPINGS BY MASTER ATTITUDE REFERENCE

EXPERIMENT TITLE	CATEGORY	REMARKS
GROUP I – STAR TRACKER OR SENSOR*		
Electrostatic Gyro (ESG)	A	Inertial
Gravity Gradient Sensor	A	Earth Reference
Earth Horizon Definition	A	Earth Reference **
Horizon Sensor Accuracy	A	Earth Reference **
Star Characteristics	B	Inertial or Earth Ref.
Gyro Compassing	B	Earth Reference
GROUP II – HORIZON SENSOR AND/OR GYROCOMPASS*		
Gravity Gradient Controls–Passive Damping	B	Earth Reference
Ion Attitude Sensing	B	Earth Reference
High Reliability Horizon Sensor	B	Earth Reference
GROUP III – NONE REQUIRED		
Low-G Accelerometer	A	Earth Reference
Gas Bearing Performance	A	
Small Impulse Devices	B	
Optical Windows and Mirrors	B	
Bearings and Lubricants	B	

*Sun tracker or sensor may be used as a supplement or, in some cases, as the prime reference.
**Gyro memory required

APPENDIX B

DATA HANDLING SYSTEMS

B.1 Introduction - The data handling system referred to in the text of the experiment descriptions includes the command link from ground to spacecraft and the telemetry data link from the spacecraft to the ground.

The basic types and complexity of the available data handling systems are defined in the second section of this appendix.

B.2 Summary of Experiment Data Handling Requirements - The review of the experiment requirements indicate that the experiment data handling requirements present only minor problems in the original design of a data handling system specifically for these experiments. The large number of data points required by some experiments may disqualify these experiments from consideration as piggyback or backup on existing spacecraft designs. Allocation of continuous channels to accommodate the high frequency data and assignment of high data rates involves equipment utilization which is not generally available for piggyback experiments on existing candidate spacecraft. Self-contained experimental packages can provide greater versatility to experiments with an increase in size, weight, and power requirements for the experiments. The completely self-contained experiment would require only a subcarrier oscillator channel and if desired the sync pulse from the master spacecraft data handling system.

B.3 Specific Data Handling Considerations - Consideration must be given to specific problems associated with the data handling equipment.

A wide variation of demands upon the data handling system is made by the various experiments as shown by Table B-1. The number of data points monitored ranges from six (five analog and one digital) for the Bearings and Lubricants

TABLE B-1
SUMMARY OF EXPERIMENT DATA PARAMETERS

EXPERIMENT	Electrostatic Gyro	Low-g Accelerometer	Gravity Gradient Sensor	Earth Horizon Definition	Horizon Scanner Accuracy	Gas Bearing Stability	Star Characteristics	Gravity Gradient Satellite Passive Dumping	Ion Attitude Sensing	Gyrocompassing	High Reliability Horizon Sensor	Star Recognition	Small Impulse Devices	Optical Windows and Mirrors	Bearings and Lubricants	TOTAL
Parameter	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	
Analog Data Points	5	5	13	21	18	19	12	7	13	8	12	13	8	14	5	173
Frequency Response (cps)	–	500	5KC	2KC	60	400	1KC	–	–	–	3	–	–	–	400	
Sampling Rate (sps)	2	1500	20	–	–	–	–	5/min.	1	1	10	1	50	1	1	
Storage – Frequency (cps)	–	500	5KC	2KC	60	–	–	–	0.1	–	3	–	–	dc	–	
Digital Data Points	6	4	5	5	5	3	4	–	–	1	1	3	–	–	1	38
Word Length (bits)	20	20	21	22	22	16	22	–	–	7	15	19	–	–	16	
Sampling rate(bps)	1	1	20	20	20	1	30	–	–	1	1	1	–	–	1	
Bit Rate (bps)	–	–	280	440	440	16	180	–	–	–	15	–	–	–	16	
Storage – Bit Rate (bps)	–	–	280	440	440	16	180	–	–	–	–	–	–	–	–	
On/Off Functions	11	–	–	2	3	7	1	6	–	1	2	–	2	13	–	48
Accuracy ≥ 1%	9	4	7	5	5	6	3	0	7	9	8	5	8	14	2	92
< 1%	0	5	11	21	18	16	13	7	6	1	5	11	0	0	4	118
Total Data Points	22	9	18	28	26	29	15	13	13	10	15	16	10	27	6	257
Spacecraft Data Points	2	3	6	10	10	–	10	–	3	6	4	11	3	–	–	68
Discrete Ground Commands	7	3	5	5	3	4	2	10	2	2	4	5	2	3	2	

Experiment to twenty-nine for the Gas Bearing Performance Experiment. Analog frequency requirements range from dc to 5 Kc. Digital bit rates vary from 7 bits/sec (bps) to 400 bps with word lengths from 6 bits to 22 bits. The number of experiments from Table B-1 which require word lengths greater than 15 bits is ten while four out of the fifteen experiments have no digital data points. Storage requirements for analog frequencies range from dc to 5 Kc and for digital bit rates range from 7 bps to 440 bps. Command system requirements shown by the table range from a minimum of two to a maximum of ten discrete ground commands.

Data from the spacecraft in terms of attitude, rates, temperature, etc. are required by eleven out of the fifteen experiments. The requirements vary from two to eleven data points. The accuracy requirements placed upon the data handling system is shown as the number of data points requiring one percent or better and those which require less than one percent accuracy. Less than one half of the data points (92 out of 257) require one percent or better.

The type of telemetry system which the carrier vehicle has onboard will determine the number and nature of the various problems. For example the Earth Horizon Definition Experiment monitors twenty-one analog and five digital data points. If this experiment is placed piggyback onboard a spacecraft which has an all PCM telemetry system the twenty-one analog points must be multiplexed and encoded into the correct format. The spacecraft data handling system must provide the experiment package with twenty-one analog channels. Otherwise, multiplexing must be provided internal to the experiment package. This arrangement requires that commutation synchronization pulses be provided by the spacecraft system to make the data available at the proper time when the experiment channel is activated. The digital data from this experiment requires five channels with word lengths up to twenty-two bits. If eight bit words are used in the spacecraft system up to fifteen digital channels could be required. Again multiplexing and parallel to serial conversion techniques employed in the experiment package will reduce the channel requirements. As with the analog data a sync is necessary to make the data available at the time required by the spacecraft multiplexing system.

The above discussion assumes one or more data channels available from the spacecraft data handling system commutated with data from other experiments or the spacecraft. An alternate approach which provides the experiment with a more versatile data handling system includes one direct telemetry channel, subcarrier oscillator, with the multiplexing, encoding, format generation, etc., being conducted internal to the experiment package. Such a system allows the experiments to combine the PCM and PAM techniques in a manner best suited to the experiment parameters. The master sync from the spacecraft used as the clock source could reduce the complexity of data reduction on the ground.

Bandwidth requirements must be considered when proposing piggyback operation on an existing spacecraft or when combining several experiments to go on a special design spacecraft. The experiments shown in Table B-1 have been designed to keep the bandwidth requirements to a minimum. Four data points have bandwidth requirements of equal or greater than 1 Kc/sec. The Gravity Gradient Sensor Experiment has one data point with 5 Kc/sec bandwidth, another experiment has two data points with 2 Kc/sec, and another has one data point with a 1 Kc/sec requirement. These requirements are not strenuous but must be taken into account when combining with other systems. In general data with bandwidth of 1 Kc/sec or greater is not sampled.

General Description of Existing Equipment - The spacecraft command link equipment includes the command receiver, antenna, decoder, and logic necessary to process the ground commands and supply appropriate signals to the respective experiment packages and spacecraft equipment.

The spacecraft data link consists of the telemetry transmitter, antenna, analog/digital (A/D) converters, storage devices, sampling, multiplexing and encoding equipment.

Telemetry Data Link - The type and complexity of the data handling telemetry system is a function of the quantity and quality of the instrumentation data to be transferred from the spacecraft to the ground station, the maximum range to the station, and the electromagnetic environment. The quantity of data or data link capacity, and the required accuracy are generally the limiting or decision parameters. Considerable effort has gone into development of techniques and equipment which reduce the bandwidth requirements, weight, and volume while increasing the data capacity and accuracy of various systems.

The various techniques are classified according to the method used to modulate the telemetry transmitter. The categories presently in the greatest use include Frequency-Modulated-Frequency-Modulation FM/FM, Pulse Amplitude Modulation (PAM), Pulse Duration Modulation (PDM), and Pulse Code Modulation (PCM).

FM/FM - This technique utilizes one or more sub-carrier oscillators (SCO) which are frequency modulated by the data signals from the transducer and signal conditioning electronics. After summation of the SCO's the composite waveform frequency modulates the telemetry transmitter. The block diagram shown in Figure B-1 is representative of such a system. The system capacity is a direct function of the number of sub-carrier oscillators used. The frequencies of the sub-carrier oscillators, as shown in Table B-2, have been standardized by the Inter-Range Instrumentation Group (IRIG). This system can accommodate up to

TABLE B-2
IRIG SUBCARRIER CHANNELS

BAND	LOWER LIMIT (CPS)	CENTER FREQ. (CPS)	UPPER LIMIT (CPS)	BAND WIDTH (CPS)	NOM. INTELL. FREQ. (CPS)
1	370	400	430	60	6
2	518	560	602	84	8
3	675	730	785	110	11
4	888	960	1,032	144	14
5	1,202	1,300	1,398	196	20
6	1,572	1,700	1,828	256	25
7	2,127	2,300	2,473	346	35
8	2,775	3,000	3,225	450	45
9	3,607	3,900	4,193	586	60
10	4,995	5,400	5,805	810	80
11	6,799	7,350	7,901	1,102	110
12	9,712	10,500	11,288	1,576	160
13	13,412	14,500	15,588	2,176	220
14	20,350	22,000	23,650	3,300	330
15	27,750	30,000	32,250	4,500	450
16	37,000	40,000	43,000	6,000	600
17	48,560	52,500	56,440	7,880	790
18	64,750	70,000	75,250	10,500	1,050
*19	86,025	93,000	99,975	13,950	1,400
*20	114,700	124,000	133,300	18,600	1,900
*21	152,625	165,000	177,375	24,750	2,500
A	18,700	22,000	25,300	6,600	660
B	25,500	30,000	34,500	9,000	900
C	34,000	40,000	46,000	12,000	1,200
D	44,620	52,500	60,380	15,760	1,600
E	59,500	70,000	80,500	21,000	2,100
*F	79,050	93,000	106,950	27,900	2,800
*G	105,400	124,000	142,600	37,200	3,700
*H	140,250	165,000	189,750	49,500	5,000

*Proposed Additional Channels

FM/FM BLOCK DIAGRAM

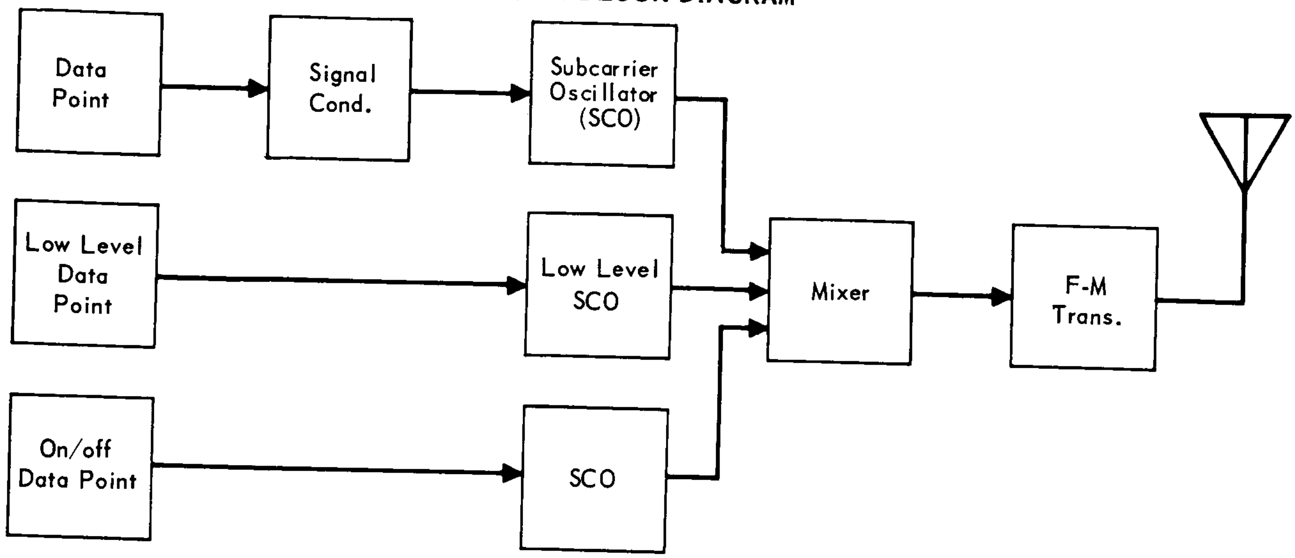


FIGURE B-1

twenty-one channels (three proposed). An additional twenty-two constant band-width sub-carriers are proposed. FM/FM telemetry systems provide simplicity in design and as a result have received the widest use. The accuracy of an FM/FM system is in the range of two to three percent.

PAM - The data handling capacity of an FM/FM system can be greatly increased by adding a commutator between several signal conditioners and a sub-carrier oscillator as shown by Figure B-2. The commutator (see Figure B-3(a)) rapidly samples the relatively slow varying data from the transducer signal conditioners and provides the sub-carrier oscillator with a series of amplitude varying pulses in a predetermined logical sequence. According to the sampling theorem, if a waveform is sampled at a rate of twice the highest frequency component, the wave-form may be reconstructed by filtering the resultant spectrum at half the sampling rate. In practice 3 to 4 samples are required. The system can handle large quantities of data, limited by the bandwidth allocation of the sub-carrier channel. The system is limited to accuracies of the same order as the FM/FM system.

PAM/PDM BLOCK DIAGRAM

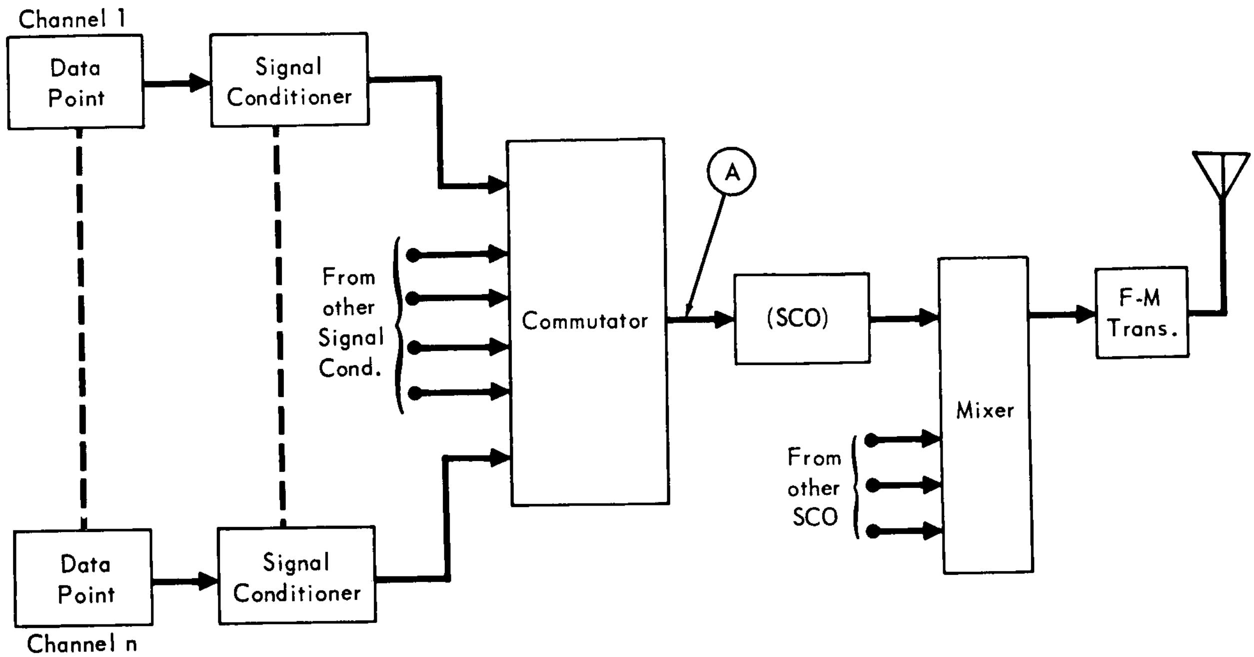


FIGURE B-2

PDM - Insertion of an amplitude-to-pulse width converter at point (A) of Figure B-2 will convert the PAM system diagram to a PDM system diagram. The commutator for the PDM system must provide the keyer synchronization and timing information in addition to sampling the data points. Pulse position modulation (PPM) is similar to PDM except amplitude to time delay conversion results in variation of the time delay between a reference pulse and a variable position pulse instead of a variation in a pulse width. Typical waveforms are shown in Figure B-3 (b and c). PDM (and PPM) is advantageous over PAM because of equipment limitations, such as drift, stability and dc response.

PCM - The previous modulation systems carried the information in an analog form, e.g., continuously variable. For PCM the information is discretely coded. Pulse code modulation systems have been under consideration since 1947 but until recently reliable equipment has not been readily available. In

COMMUNTATOR SAMPLING

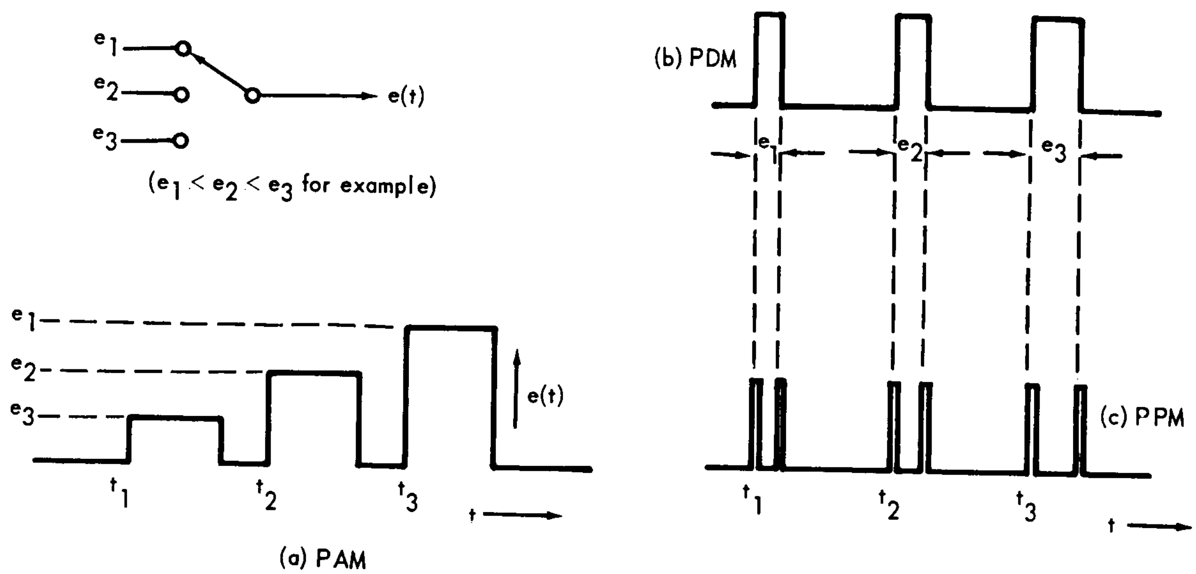


FIGURE B-3

the past five years PCM technique has surged forward to become a standard on several space programs.

In the PCM system the transducer signal conditioner output is quantized into discrete levels each represented by a distinct code group. The presence or absence of a pulse or series of pulses make up the code group.

A block diagram of a typical PCM system is shown in Figure B-4. The transducer data is sampled as in the previous systems by a high-speed commutator, or multiplexer. The output of which is a series of amplitude varying pulses. These pulses are sent to a high-speed analog to digital converter where the magnitude of the input pulses is represented by a coded pulse group at the converter output. This entire process is generally referred to as data encoding. The coded output of the encoder is usually binary, 000···000 for zero input level and 111···111 for full scale. Gray codes and other special codes have been developed for transmission of information. The accuracy of the PCM system can be held to ± 1/2 the bit quantizing level if the signal received at the ground station is

PCM BLOCK DIAGRAM

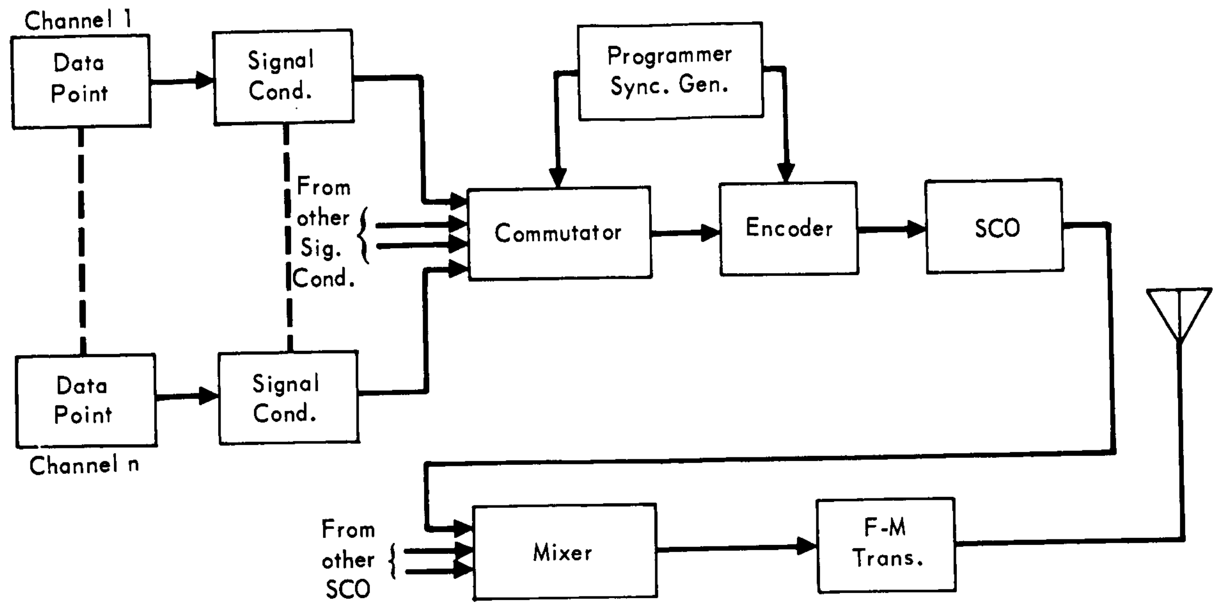


FIGURE B-4

well above the receiver threshold, which is usually 15 - 16 db for PCM. Error detection and correction techniques may be used to upgrade the received data. These techniques have received considerable attention in technical papers and it is not within the scope of this appendix to discuss the relative merits of various schemes. It will suffice to say that usually such techniques require greater bandwidth, for example, the simplest error detecting scheme designates one or more bits of the code group (word) as a parity bit to allow the receiver detection logic to recognize single bit errors.

For simplicity it is generally desirable to keep the digitally originated and the analog originated data on separate systems with constant word lengths. The PCM systems have the greatest complexity of those in common use, however, the high accuracy, capacity, and versatility of the PCM system has led to the increasing use of the system for complex space applications.

<u>Associated Data Handling Equipment</u> - The previous paragraphs have mentioned a wide variety of equipment. In addition to the aforementioned equipment, data

storage is required to preserve the information generated while the spacecraft is out of range of the ground station. Magnetic core memory elements, magnetic tape storage, and semiconductor bilevel elements have been used in various combinations to obtain the required storage capacity.

Magnetic tape storage featuring either or both analog and digital recording modes has been used effectively on recent major space vehicles. Two of the observatory satellites and both long term manned space vehicles propose the use of magnetic tape storage for data collection. One of the orbiting observatories will use magnetic core-type data storage, the primary advantage of core memory being that of reliability through the elimination of moving parts. The tape storage sys-system, however, can provide a higher storage capacity per unit volume and can be used in an all analog system. The storage system can be utilized to affect data compression or expansion. For example, data recorded at a slow recorder speed, and with subsequent playback at a higher speed, can result in a data and time compression of typically 26 to 1; then data recorded for 260 minutes could be dumped during a 10 minute pass over a ground station. The amount of compression which can be used is a function of the spacecraft and ground system bandwidth capability, the allowable error which increases with increasing data rate, and the capability of the particular tape unit. Data expansion involves the reduction of the data rate (and bandwidth) during the playback mode by reducing the playback speed. This feature would be utilized to reduce the bandwidth requirements of the spacecraft and ground systems, particularly for deep space missions.

The pacing parameters in the utilization of magnetic tape storage systems are the accuracy and the capacity. For an analog system these are equivalent to a percent tolerance of reproduceability of the original signal and the frequency response of bandwidth respectively, and the parameters, for the digital system, correspond to the number of bits per one error bit and the number of bits per inch (packing density). Both accuracy and packing density or frequency response have received a considerable amount of research in recent years. Current bit packing densities range from 100 to 2000 bits/inch (bpi). Advanced experimental designs indicate practical densities of 100,000 to 200,000 bpi. Table B-3 indicates various parameters of typical satellite recorders. Number of channels, tape length and speed ranges in most systems must be selected for the particular application.

TABLE B-3

TYPICAL DATA HANDLING EQUIPMENT

DEVICE	MFR.	PROJECT USE	FREQ. BAND	VOL. (CU.IN.)	INPUT POWER (WATTS)	OUTPUT POWER (WATTS)	CHANNELS	BIT PACKING DENSITY (BITS/IN.)	TYPE MODULATION	WEIGHT (LB.)
Data Storage (Magnetic Tape)	RCA	Gemini	–	347	15	–	5 pcm 9 Analog	640	Digital & Analog	15
	Leach	Apollo	–	975	35	–	4 pcm 9 Analog 1 Clock	853	Digital & Analog	30
	Borg-Warner	–	–	560	35	–	8 Digital	500	Digital	15
T-M Transmitter	RCA	Gemini	VHF	36	17	2	–	–	FM	2
	UED	–	VHF	23	–	2.5	–	–	FM	1.4
	Conic	–	VHF	29	25	5	–	–	FM	1.6
	Conic	–	S or L	31	70	5	–	–	FM	3
	ECI	–	S	47	–	3	–	–	FM	2.6
Subcarrier Oscillator	UED	–	–	2.1	0.6	–	1–18/A-E	–	FM	2.5 oz.
Analog-Digital Converter	UED	–	–	44	3.8	–	9 bits	–	–	2.7
Transponder	ITT	–	S	60	5	0.05	–	–	CW-Phase	2
	Motorola	Saturn	C	100	25.2	700	–	–	Pulse	5.5
Command Receiver	Motorola	–	UHF	26	2	–	3	–	Tone	2
Multiplexer Encoder	AEC	–	–	240	17	–	31–127	–	Binary	11

Multiplexing of several data sources onto one telemetry channel was previously mentioned as one method of increasing the data handling capacity of the telemetry system. A commutator, either mechanical or solid state, is used to time share the data channels. Digital data systems may combine commutation and analog-to-digital encoding into a multicoder unit. A typical PCM multicoder is shown in Table B-3. Units are also available to provide pulse duration modulation (PDM) outputs from a multiple channel input. Typically the multiplexing units operate with a 0 to 5 volt input, however, several units are available which will accommodate low level differential signals of 0-5 mv or 0-50 mv. In the most straightforward systems the multiplexed signal in any of the forms (PAM, PDM, PPM, PCM) is used to modulate the transmitter directly by frequency or phase modulation of the primary oscillator, for example, PAM/FM or PAM/PM. The system capacity is increased by using more than one commutator, each modulating a sub-carrier oscillator (SCO); for example, PAM/FM/FM or PAM/FM/PM. The potential capacity of the system is limited by the bandwidth allocations.

The sub-carrier oscillators may receive input signals from the multiplexer, data storage, or directly from the data transducer. These units are generally voltage controlled oscillators (VCO) and are available for both high-level, single-ended and low-level, double-ended signals. Oscillators are also available which are current or reactance-controlled. The oscillators are usually confined to the IRIG frequencies previously listed in Table B-2; special frequencies can be obtained.

Telemetry transmitters are available which cover the entire spectrum. Current missile/space allocations are 216-260 mc, 1435-1535 mc, and 2200-2300 mc. The VHF band (216-260 mc) will be phased out of the space communication network by 1970. Typical power output of 0.5 to 20 watts is available. Special powers of a few milliwatts to kilowatts are also available. Power output should be restricted to the minimum required for the application.

Command Data Link - The ground to spacecraft command data link provides the following functions:

a. Sequencing of functions when no automatic programmer is onboard.

b. Override automatic programmer in case of failure or unusual data.

c. Request data; either (or both) real time or stored.

d. Request tracking or other functional aids.

The command receiver is generally a superhetrodyne receiver designed to handle tone, phase shift, or frequency shift modulation. A typical receiver with three tone channel capability is included in Table B-3. The command data may be coded in the form of digital words to secure the command link and/or to increase the data capacity. The command decoder and logic network associated with the receiver perform the decoding and distribution functions. The number of discrete ground commands required for the spacecraft mission, range from a relative few to several hundred. The number of commands which can be processed during a single contact between the ground and spacecraft is a function of the data word length, rate, processing logic, command data storage and of course, the duration of the contact.

B.4 Conclusions - Several conclusions can be drawn from the discussion of the experiment requirements and the equipment and techniques available for data handling.

a. Data handling requirements for the experiments are compatible with off-the-shelf and state-of-the-art data handling equipment.

b. The existing and proposed ground station data processing equipment, telemetry, command, and post flight capability is adequate for the proposed experiments.

c. Experiments which require accuracies of one percent or better of full scale for analog data will usually require digital data handling. In some cases, where the one percent accuracy was dictated by the resolution or accuracy at small signal level, dual range instrumentation will provide signals compatible with an analog data system.

d. The majority of experiments can use a three-channel command receiver (seven discrete commands). Those with eight to fifteen commands require a four channel system.